STAT

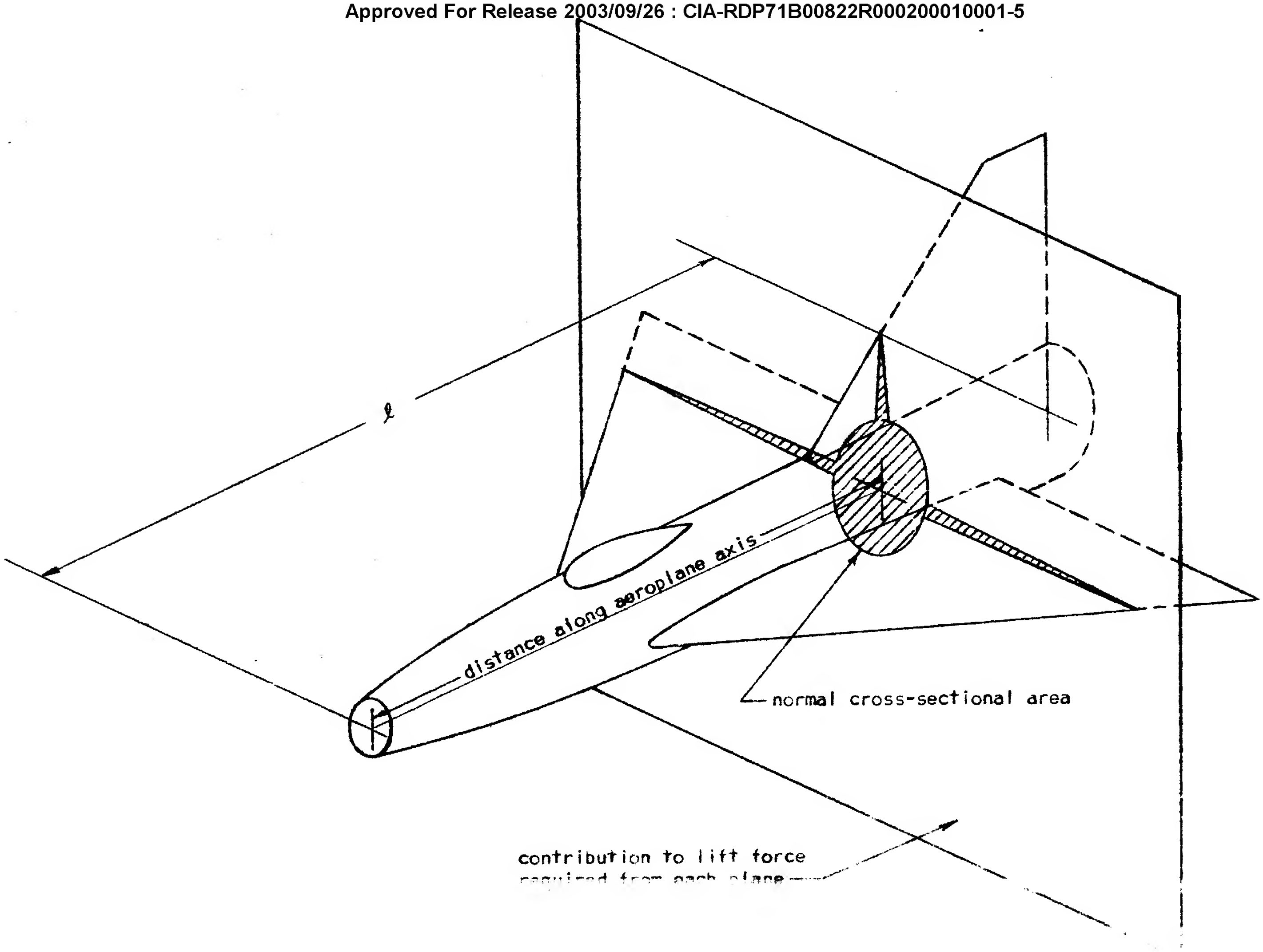
In order to produce the detailed wave structure and the magnitude of bang vertically below a supersonic aircraft flying in a non-uniform atmosphere, one needs the following data:

1. Flight Mach number and altitude
2. Configuration of the aeroplane
3. Normal cross-sectional area distribution of the aeroplane as a function of distance along aeroplane axis. This area includes the fuselage, wings, and nacelles. See attached Figure.
4. For any flight Mach number of interest, one needs the lift distribution as a function of distance along aeroplane axis. This lift distribution is obtained by integrating laterally the pressure forces on the lifting surfaces. See attached Figure.
5. Range of cruising $C_L AR$.

   $C_L AR$ is defined as

$$\frac{W}{\frac{1}{2}\rho_h U^2 l^2}$$

   where l is the characteristic length of the aeroplane.


ℓ
distance along aeroplane axis
normal cross-sectional area
contribution to lift force

A NOTE ON THE PRESSURE DISTURBANCE AT GROUND LEVEL CAUSED BY HIGH-FLYING SUPERSONIC AIRCRAFT

by
N. C. Freeman and S. H. Lam

Princeton University

AFOSR TN 58-1127
Report No. 444

AD 207 781
December, 1958

UNITED STATES AIR FORCE

Office of Scientific Research
Air Research and Development Command

Contract AF 49(638)-465
Project No. 9781

SUMMARY

This brief report studies the pressure disturbance or bang produced on the ground by a high-flying supersonic aircraft. The main interest here is: (1) to assess the effect of the non-uniformity of the atmosphere in attenuating the waves from the aircraft and (2) to relate this to some gross parameters of the aircraft.

The linear supersonic flow theory is used to compute the pressure disturbance at large distance from the aircraft and the attenuation of this disturbance through a Rand standard atmosphere is computed by a simple application of Whitham's theory. Formulae for either a "lift-controlled" or "shape-controlled" bang are given as a function of the gross parameters of the aircraft such as flight Mach number, altitude, size, slenderness ratio, lift coefficient and certain design factors which depend on rather detailed knowledge of the aircraft in question.

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ACKNOWLEDGEMENT

The present study is part of a program of theoretical and experimental research on high speed gasdynamics ~~viscous effects in supersonic flow~~ being conducted by the Gas Dynamics Laboratory, The James Forrestal Research Center, Princeton University. This research is sponsored by the Office of Scientific Research, Air Research and Development Command, Fluid Mechanics Division under Contract AF 49(638)-465.

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A NOTE ON THE PRESSURE DISTURBANCE AT GROUND LEVEL CAUSED BY A HIGH-FLYING SUPERSONIC AIRCRAFT

The pressure field associated with a supersonic lifting body can be obtained from the theories of linearized supersonic flow of Hayes[1] and Ward[2] together with the extensions to include non-linear effects at large distances given by Hayes[3] and Whitham[4]. The introduction of a non-uniform variation of the atmosphere is then straightforward as noted by Whitham. A recent extensive investigation by Struble et. al. treated this problem but their adaption of the Whitham technique to consider the effects of atmospheric variation appears to be incorrect.

The difficulties encountered in the present note stem not from a development of the above theory, but from an attempt to interpret its results in terms of "gross" parameters of the aeroplane. Given all the details of area distribution, lift distribution, etc., it is possible to obtain, within the limits of the above theory, a precise picture of the detailed flow at large distances. The complicated non-linear interaction of the pressure waves produced by the "lift"-induced and the "shape"-induced pressure fields makes a general discussion of the behavior under arbitrary values of all the relevant parameters impossible. Indeed, it may be deduced from this difficulty that the design of the aeroplane itself has a particularly important place in the size of the disturbance produced as the aeroplane flies at different altitudes. If a small pressure disturbance were one of the design criteria of the aeroplane, it would seem possible to arrange this. Whether such an aeroplane would be acceptable from other points of view remains doubtful.

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In general, it is possible to extract an explicit form for the variation of the pressure at ground level with height for a particular aircraft flying at a fixed Mach number. The difficulties are associated with comparing two different aircraft or the same aircraft flying at two different Mach numbers, if we limit outselves to a requirement that the result be in terms of the "gross" parameters of the airplane.

For a given aeroplane flying at a given Mach number, the linearized wave pattern far away from the body and near the Mach cones can be calculated. This flow pattern serves as "initial conditions" in considering the non-linear attenuation of the waves. Without a detailed knowledge of the configuration and lift distribution of the aeroplane, one could not describe the process of interaction of the wave patterns. However, asymptotically, the wave pattern will tend to a single N-wave (except in very special circumstances where more than one N-wave may persits), and it is the strength of the head shock of this N-wave that the analysis predicts.

The linearized pressure fields due to the lift and due to shape of the given aeroplane may interfere in different ways at different flight Mach numbers. The interaction controls the strength of the head shock. In the present theory, one assumes that the bang heard on the ground is controlled by either the "lifting" pressure field or the "shape" pressure field. In other words, one considers the limiting case of when either pressure field is so much more powerful than the other one that the interference effect is negligible.

For the lift-controlled bang, the pressure rise across the head shock is

$$\frac{\Delta p}{p_g} = \frac{\gamma}{2^{3/4}\sqrt{\pi}} \left(\frac{M_g^2}{M_h} \ell \right)^{3/4} \sqrt{C_L \mathcal{R}} \quad \frac{Z(h)}{h^{3/4}} \quad d_L(M_h) \qquad (1)$$

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and for a shape-controlled bang, the pressure rise across the head shock is

$$\frac{\Delta p}{p_g} = \frac{\gamma}{2^{1/4}}\left(\frac{R}{\ell}\right)\left(\frac{M_g^2}{M_h^{5/3}}\ell\right)^{3/4}\frac{Z(h)}{h^{3/4}}\,d_S \qquad (2)$$

where p_g is the pressure at ground level, M_h is the flight Mach number, M_g is the flight velocity divided by the speed of sound on the ground, ℓ is a characteristic dimension, d_L and d_S are the design factors (d_L is a function of flight Mach number and d_S is a constant for a fixed aeroplane. See Appendix.), h is the altitude of flight, $Z(h)$ is a function of the atmosphere and is computed for the Rand standard atmosphere and plotted in Figure 1. The factor R/ℓ is the slenderness ratio of the cross-sectional area of the aeroplane. The factor C_LAR is defined as

$$C_LAR = \frac{2W}{\rho_h U^2 \ell^2}$$

where W is the weight of the aeroplane, ρ_h is the density at flight altitude and U is the flight velocity.

In formulae (1) and (2), the approximation $M \doteq M^2-1$ has been made, where M is the flight velocity divided by the local speed of sound at any altitude below the aeroplane. Since the speed of sound variation for the standard atmosphere is only about 10% for altitudes below 80,000 feet, one could replace M_g by M_h in the formulae without making any serious error. Thus, formulae (1) and (2) can be written as:

Lift-controlled

$$\frac{\Delta p}{p_g} = \frac{\gamma}{2^{3/4}\pi^{1/2}}\,(M\ell)^{3/4}(C_LAR)^{1/2}\,\frac{Z(h)}{h^{3/4}}\,d_L(M) \qquad (3)$$

Form-controlled

$$\frac{\Delta p}{p_g} = \frac{\gamma}{2^{1/4}}\left(\frac{R}{\ell}\right)M^{1/4}\,\ell^{3/4}\,\frac{Z(h)}{h^{3/4}}\,d_s \qquad (4)$$

In principle, if the configuration and lift distribution of the aeroplane are known in detail, the progress of the interaction of the linearized wave patterns can be predicted. The asymptotic head shock strength would then be given by

$$\frac{\Delta p}{p_g} = \frac{\gamma}{2^{3/4}\pi^{1/2}} (M\ell)^{3/4} \sqrt{C_L AR} \; \frac{\varepsilon(h)}{h^{3/4}} \; d_{LS}\left(M, \left(\frac{R}{\ell}\right)^2 \frac{1}{\beta_h C_L AR}\right)$$

where d_{LS} is again a design factor which is quite a complicated function of the configuration and lift distribution of the aeroplane. The detailed definition of d_L, d_S and d_{LS} will be given in the Appendix. It will suffice to state here that they are of order unity.

It is seen from formulae (3) and (4) that no single parameter appears to dominate either result.

It is obvious that the effect of altitude is to decrease the strength of the bang. The lift-controlled bang is discussed first. To increase altitude for an aeroplane of a given weight and flight Mach number would necessitate an increase in C_L, which tends to increase the bang. Figure 1 also shows the overall effects of altitude for this case. It is seen that the bang still decreases with altitude, but more gradually.

For a given shape of the aeroplane and a given flight Mach number, the scale of the aeroplane is indicated by the length ℓ. If $C_L AR$ is kept constant, the bang is directly proportional to $\ell^{3/4}$, and if W is kept constant (since ARC_L is non-dimensionalized by a factor ℓ^{-2}), the bang is inversely proportional to $\ell^{\frac{1}{4}}$. This shows that for a given total lift, the more spread out the lift distribution is the less bang it makes on the ground.

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If the bang is shape-controlled, then for a given shape of the aeroplane, formula (4) gives explicit Mach number dependence. The interpretation of this formula is then explicit.

In general, both the lift-controlled and the shape-controlled pressure fields will be headed by a compression phase in the groundward direction. Thus, the two pressure fields will tend to reinforce each other as they interact. Therefore, by computing the lift-controlled and shape-controlled disturbances by formulae (3) and (4), the larger of the two will give the minimum bang produced by the aeroplane under the most facorable interference conditions.

It must be realized that these results are also subject to the limitations of the theory itself. Damping effects due to viscosity outside the shock waves and reflections at local temperature discontinuities are not included. Both these effects might tend to reduce the pressure variation. It would, however, be possible to think of a temperature variation which could focus the waves and, hence, produce a locally very large pressure disturbance.

As the thickness of a shock wave varies inversely with its strength, the thickness of the head shock wave would become comparable to the size of the N-wave itself at very large distances. A more refined theory is then needed to consider this effect.

EXAMPLE:

	Aeroplane A	Aeroplane B
M	2	3
R/	Same	Same
ℓ	1	2
h	50,000'	70,000'
C_L	Same	Same
AR	Same	Same

Then for the "lift"-controlled pressure

$$\frac{\Delta p_A}{\Delta p_B} \simeq 0.9$$

and for the "shape"-controlled pressure

$$\frac{\Delta p_A}{\Delta p_B} \simeq 1.2$$

where we have assumed d_S and d_L to be the same for the two aircraft.

In view of the approximations used in the theory, it would not be realistic to take variations of less than 20% as significant. In general, we would expect the lift-controlled pressure to be most important. Thus, the result would seem to indicate that Aircraft B will give about the same disturbance as Aircraft A.

In general

$$d_{LS} = \sqrt{\int_0^{\tau_{LS}} \left\{ F_L(\tau) + \frac{R^2}{\ell^2}\frac{2\pi}{\beta_h C_L AR} F_s(\tau) \right\} d\tau}$$

where F_L and F_S are the disturbance functions associated with the lift distribution and area distributions. The limit of the integrat on τ_{LS} is associated with the detailed wave pattern of the linearized flow field (see Whitham[4] and Lighthill[5] for a more detailed discussion of this point).

F_L is defined as

$$F_L = \frac{\ell}{W}\int_0^{\tau} \frac{\frac{dL}{dt}\,dt}{\sqrt{\tau - t}}$$

where L is lift per axial distance and F_S is defined as

$$F_S = \frac{1}{\pi R^2}\int_0^{\tau} \frac{\frac{d^2S}{dt^2}\,dt}{\sqrt{\tau - t}}$$

where S is cross-sectional area of the aeroplane. Both F_L and F_S are non-dimensionalized quantities. For different aeroplanes of very similar design, one might expect the corresponding design factors to be roughly the same. This is the justification for assuming d_L and d_S to be the same in the given example.

The two limiting cases considered correspond to the two terms in the integrand:

$$d_L = \sqrt{\int^{\tau_L} F_L\, d\tau} \quad ; \quad d_S = \sqrt{\int_0^{\tau_S} F_S\, d\tau}$$

In these cases τ_L and τ_S denote the first zeros of F_L and F_S, respectively.

The altitude function $Z(h)$ is

$$Z = \sqrt{\frac{P_h}{P_g} \frac{\sqrt{h}}{\sqrt{\rho_1 a_1^5} \int_0^h \frac{1}{\sqrt{\rho a^5}} dr}}$$

where a is the speed of sound.

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7. Grimminger, G.: Analysis of Temperature, Pressure, and Density of the Atmosphere Extending to Extreme Altitudes. Rand Report 105, November 1, 1948.

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4. Personal authors: N. C. Freeman and S. H. Lam

5. Date of Report: December, 1958

6. Pages: 9

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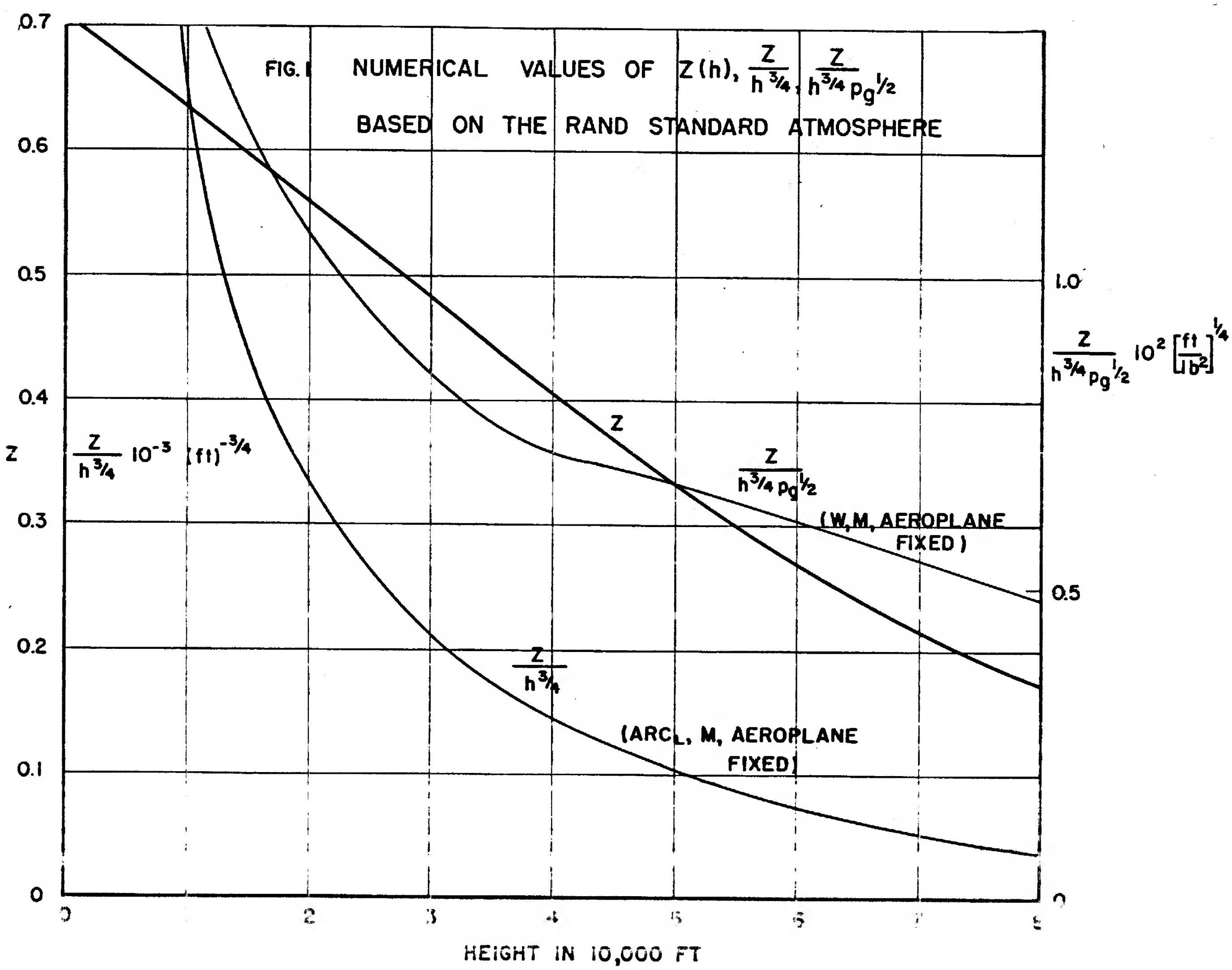
FIG. I NUMERICAL VALUES OF Z(h), Z/h^3/4, Z/h^3/4 pg^1/2
BASED ON THE RAND STANDARD ATMOSPHERE
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
Z
Z/h^3/4 10^-3 (ft)^-3/4
1.0
0.5
0
Z/h^3/4 pg^1/2 10^2 [ft/lb^2]^1/4
Z
Z/h^3/4 pg^1/2
(W, M, AEROPLANE FIXED)
Z/h^3/4
(ARC_L, M, AEROPLANE FIXED)
HEIGHT IN 10,000 FT

Figure 1.

PD-3558

The Supersonic BANG

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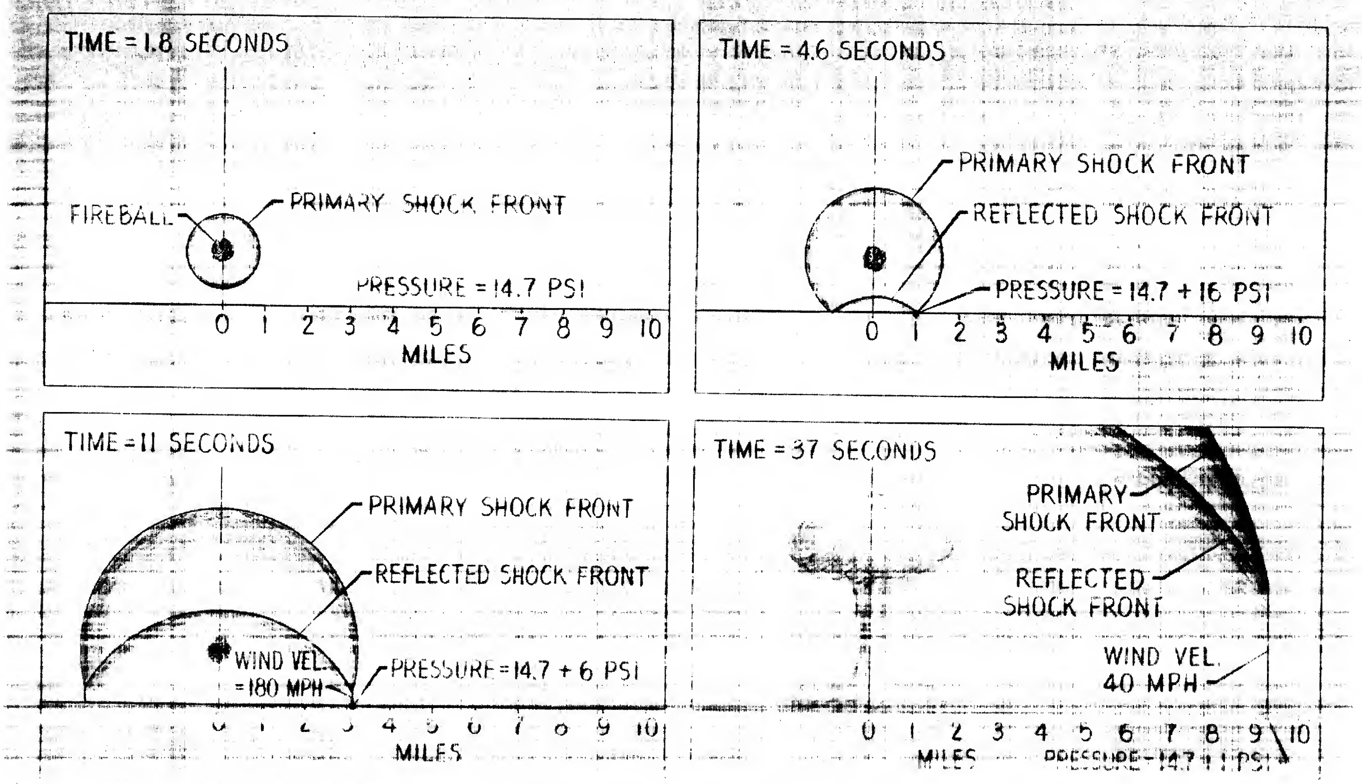
PD-3458

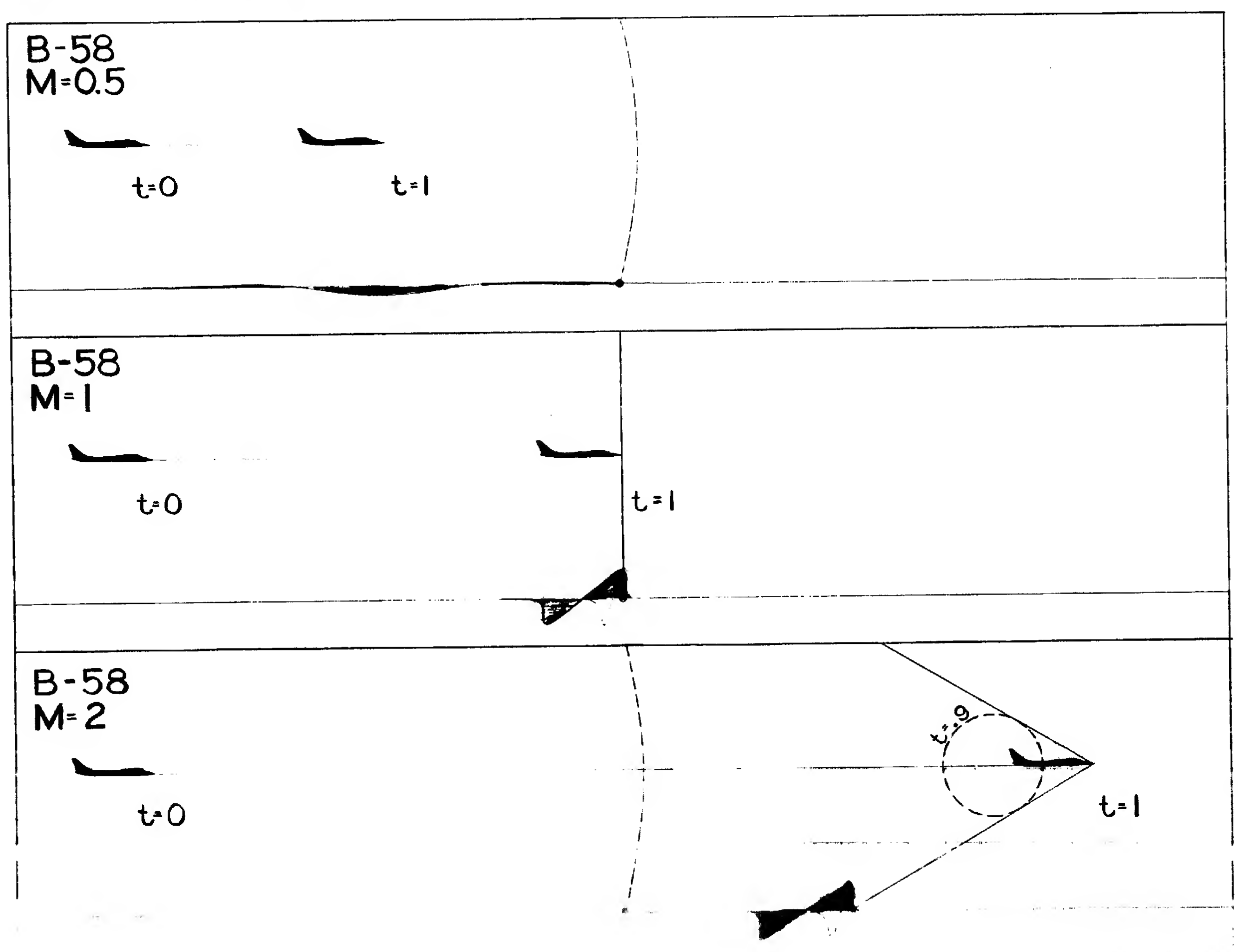
THE BIGGEST BANG — A NUCLEAR EXPLOSION!

SHOCK PROGRESSION — 1 MEGATON AIR BURST



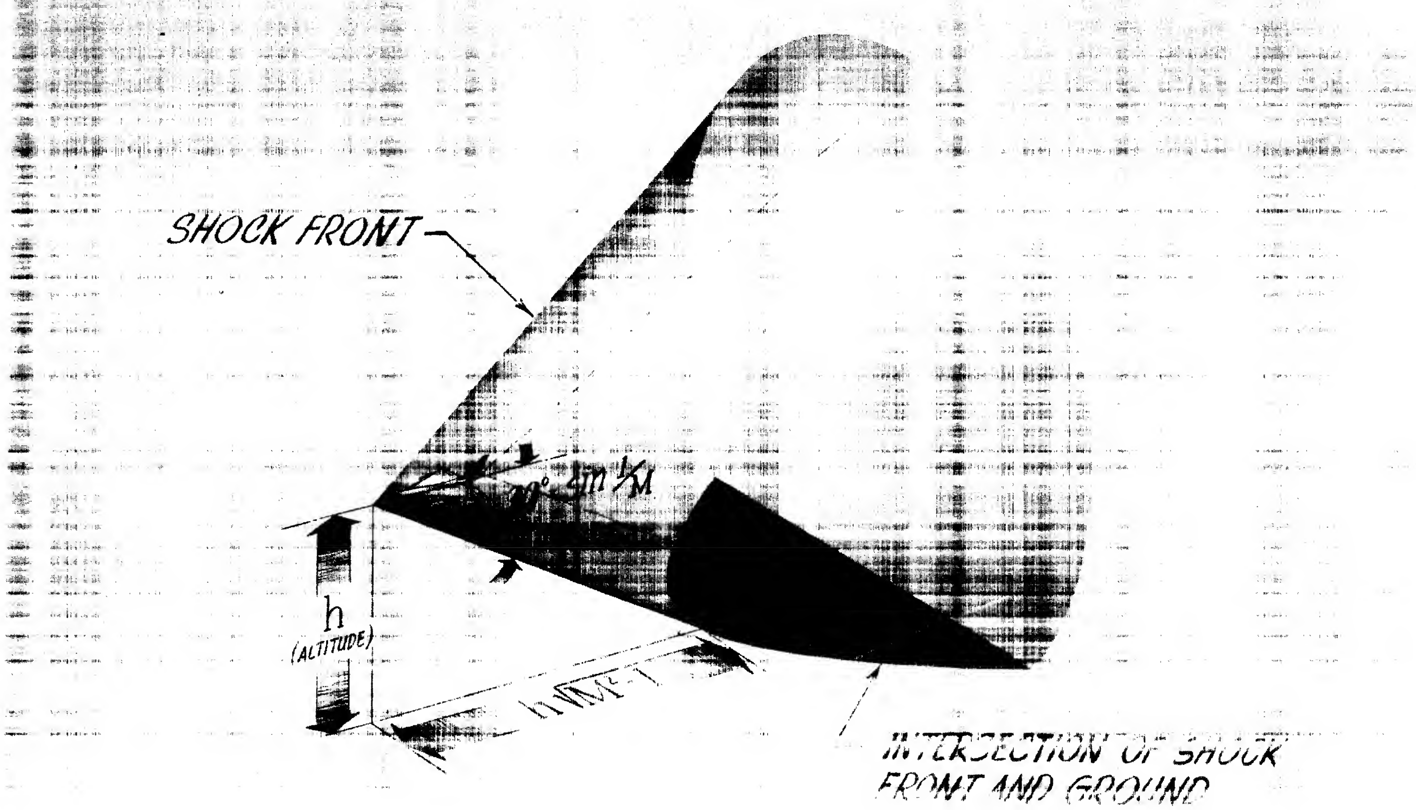
PD-3459

TIME HISTORY OF THREE AIRPLANES

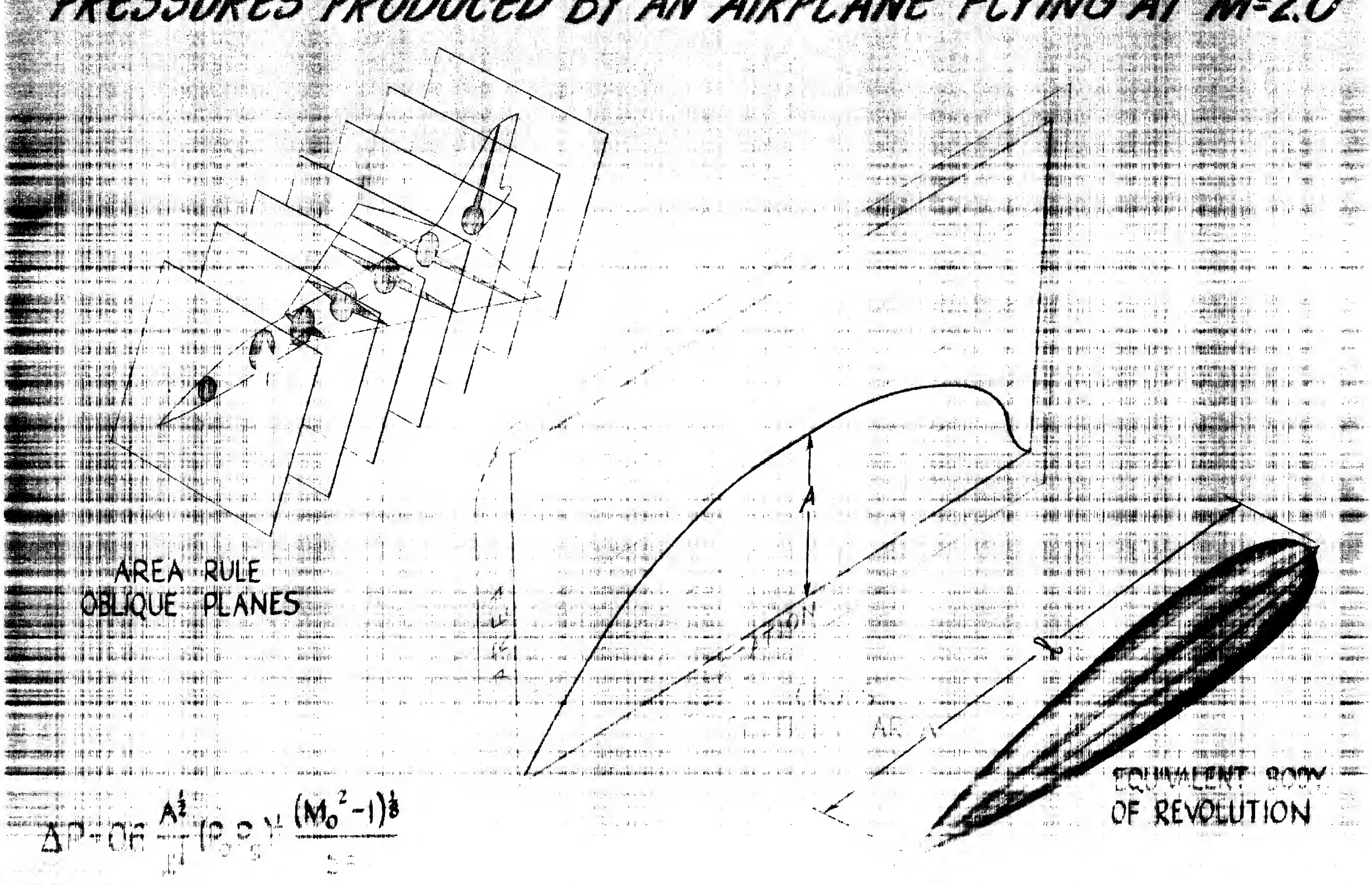


PD-3460

GEOMETRY OF THE SHOCK FRONT OF AN AIRPLANE FLYING AT MACH 2



A METHOD OF CALCULATING APPROXIMATE DISTANT PRESSURES PRODUCED BY AN AIRPLANE FLYING AT M=2.0



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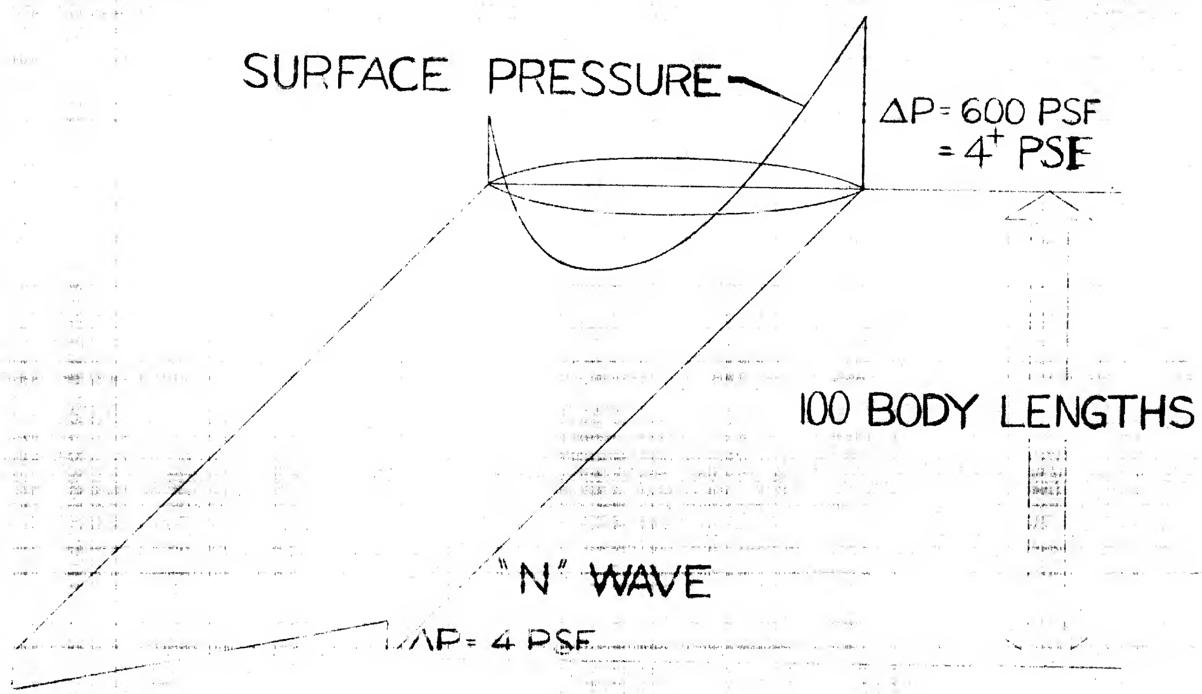
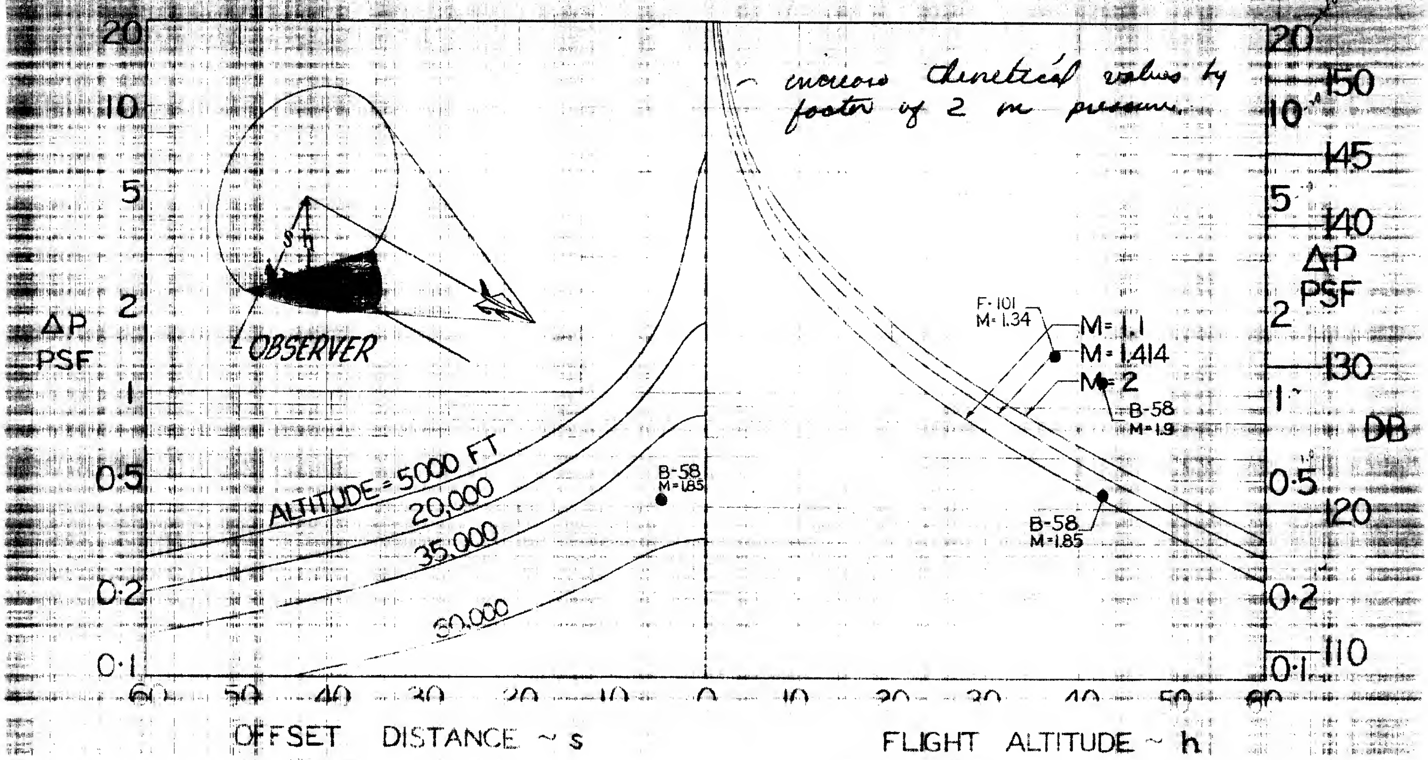
PD-3493

TYPICAL PRESSURE PATTERNS FROM A BODY OF REVOLUTION FLYING AT MACH 2

FINENESS RATIO = 9



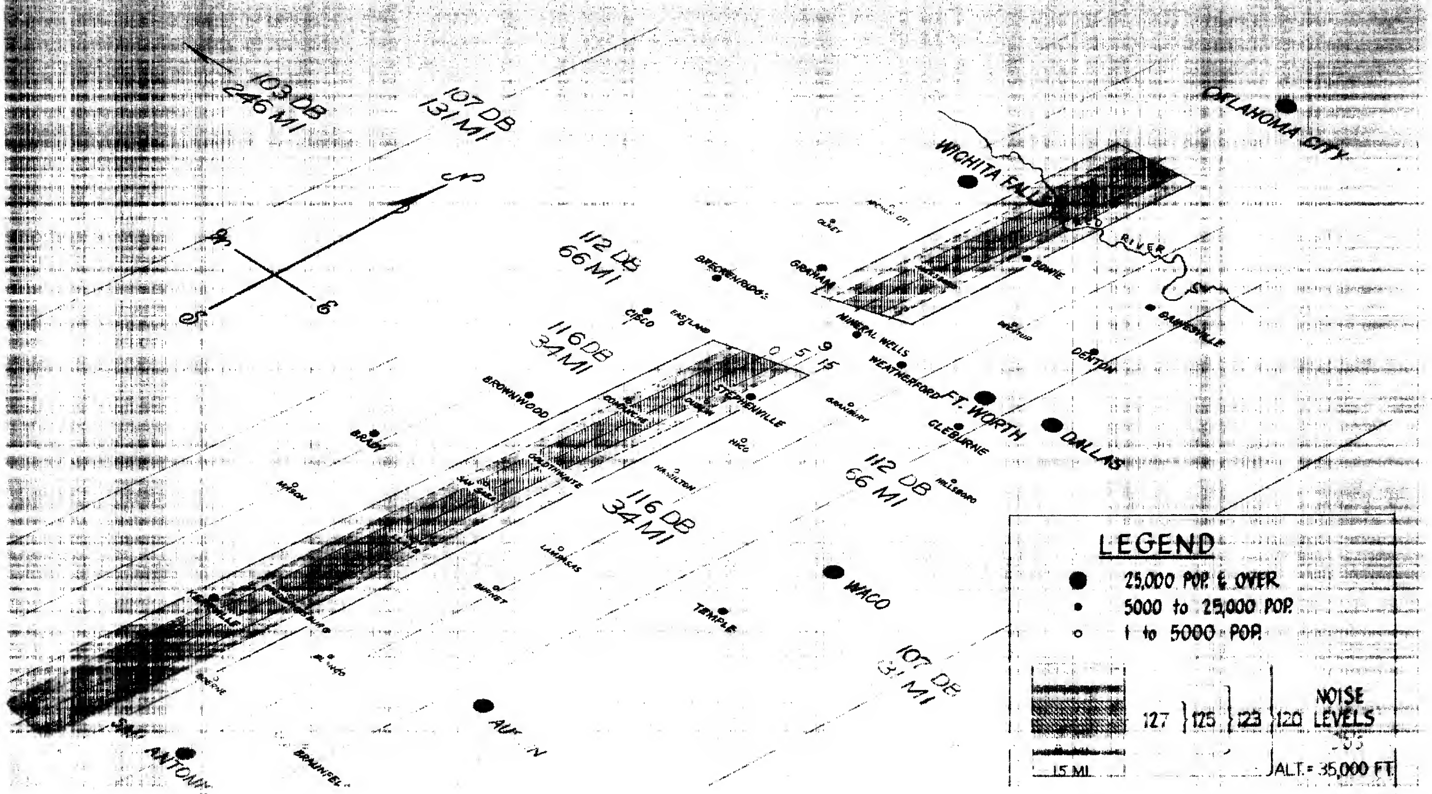
SOME SAMPLE CALCULATIONS OF THE SONIC BANG EMANATING FROM A SUPERSONIC AIRPLANE



80-3463

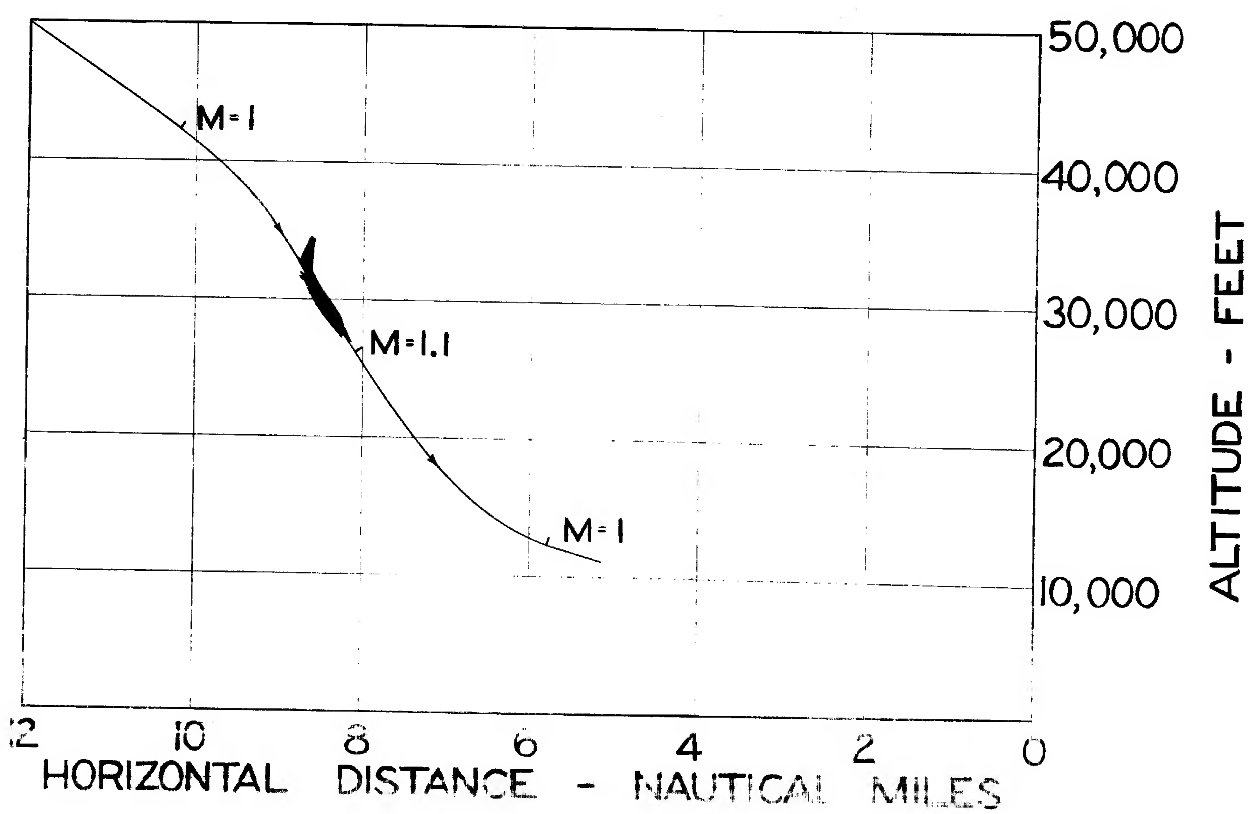
SONIC BANGS & THE AFFECTED LOCAL GEOGRAPHY

ALONG CONVAIR'S FLIGHT CORRIDOR



PD-3464

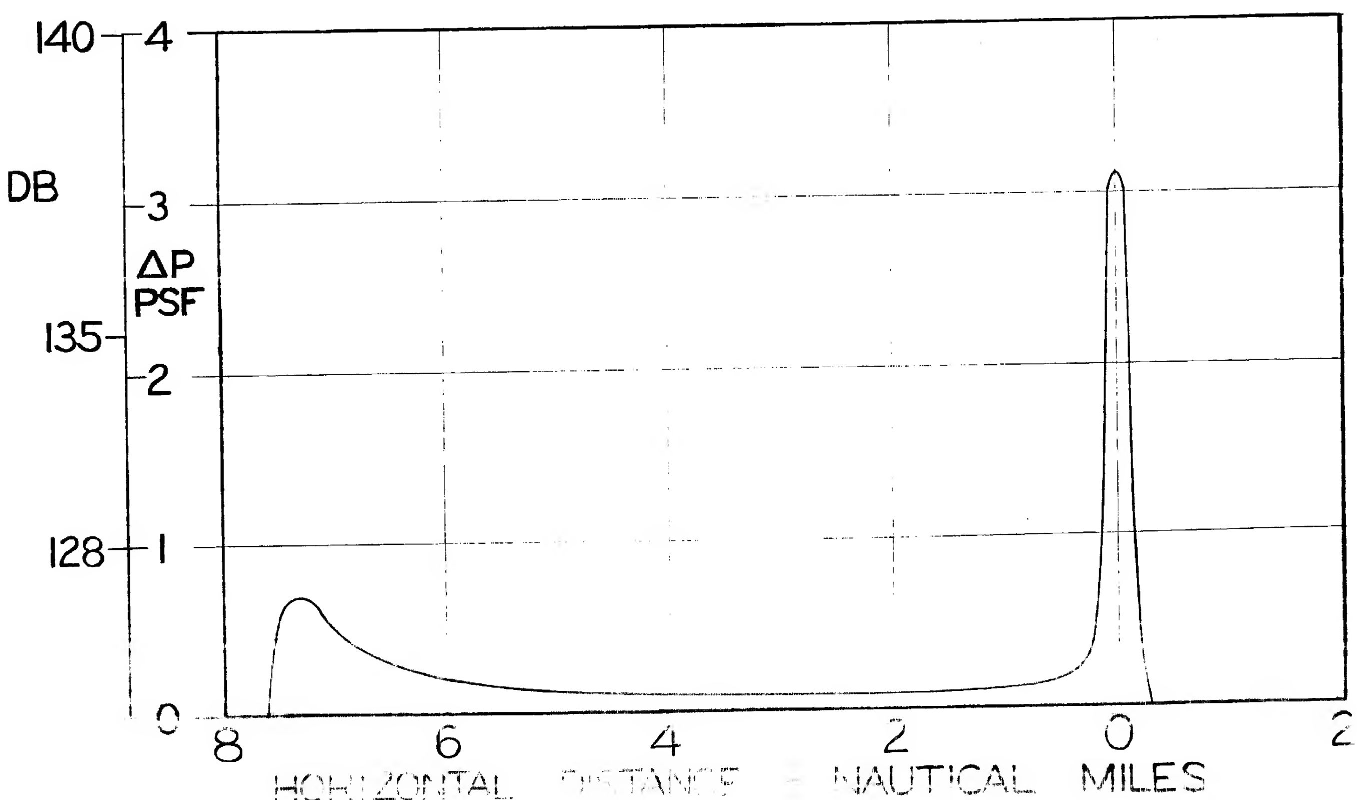
A SPECIAL CASE
ACCELERATED FLIGHT PATH

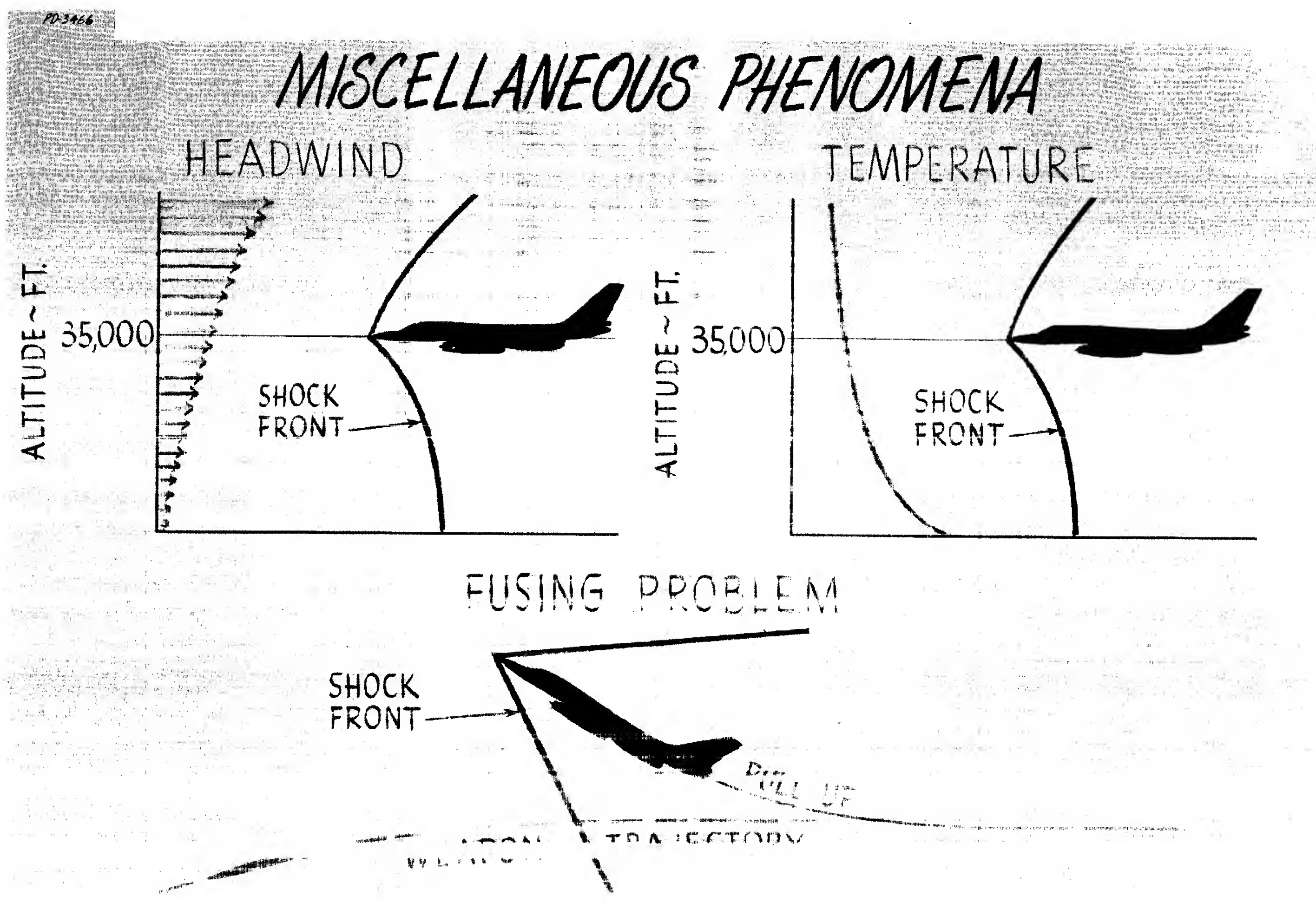


PD-3465

A SPECIAL CASE

THE PRESSURE PATTERN




PD-3466
MISCELLANEOUS PHENOMENA
HEADWIND
ALTITUDE ~ FT.
35,000
SHOCK FRONT
TEMPERATURE
ALTITUDE ~ FT.
35,000
SHOCK FRONT
FUSING PROBLEM
SHOCK FRONT

25X1

PROPOSED TEST PROGRAM

FOR

"SONIC BOOM" RESEARCH

Approved For Release 2003/09/26 : CIA-RDP71B00822R000200010001-5

A
PROPOSED TEST PROGRAM
FOR
"SONIC BOOM" RESEARCH

I. Introduction

The past two years of B-58 flight operation have emphasized the importance of the sonic boom problem. These two years have created an acute awareness among industry and military service members alike of the adverse public relations arising from booms.

The boom problems facing USAF are going to increase rapidly in the next few years. Beginning with the long sustained supersonic cruises of the B-58 in its flight test phase the frequency of public complaints has risen sharply. Later this year the B-58 enters its operational life in the Strategic Air Command. A second upsurge of complaints is expected when training flights are commenced over heavily populated areas such as are scheduled in SAC's operations.

The problem will be further accentuated by the introduction of the B-70 into flight test and operational phases. And finally, the arrival of military and commercial supersonic transports will irrevocably focus the public's attention on the sonic boom. The variety of new aircraft shapes, sizes, and performance characteristics will add more complexity to the intensity and frequency of booms. Ranging from small interceptor missiles of the Bomarc type to the huge long range bombardment and transport craft flying at hypersonic speeds, the sources and numbers of booms will continually expand. The wide range of flight Mach numbers and altitudes results in still more complexity from the viewpoint of sonic booms.

Of obvious importance to both the designer and operator of supersonic aircraft is the pressing need of a reliable method of predicting sonic boom intensities, distributions, and ground coverage. While a number of reports dealing with analytical prediction methods have been published recently*, very little effort has been expended to experimentally verify these predictions. Aside from limited comparisons of theoretical and experimental boom results made by this Division, Convair knows of no integrated evaluations of such value as to permit confident predictions of new aircraft.

The Langley Research Center of NASA has conducted some model and flight test measurements of booms but not in an integrated program as is required. Accordingly Convair proposes the combined wind tunnel and flight test program of sonic boom investigation as is embodied in this report. It is our opinion that this program is needed to provide reliable boom estimates of present day and future high performance aircraft.

*For example, see AFOSR TN 58-1127 Report No. 444 (AD 207781) prepared by the Gas Dynamics Laboratory of Princeton University.

II. The Composition of Sonic Booms and a Recommended Course of Action

Existing theoretical methods for predicting "sonic boom" intensities have been surveyed, exploited, and compared with existing experimental data. A breakdown of a sonic boom with its ingredients is shown in Figure 1.

The intensity of a sonic boom is measured as overpressure and is seen to be the product of five basic parameters, each of which in turn is affected by one or more factors. The successful prediction of the sonic boom intensity created by a supersonic vehicle is dependent upon the experimental verification of each of these five parameters.

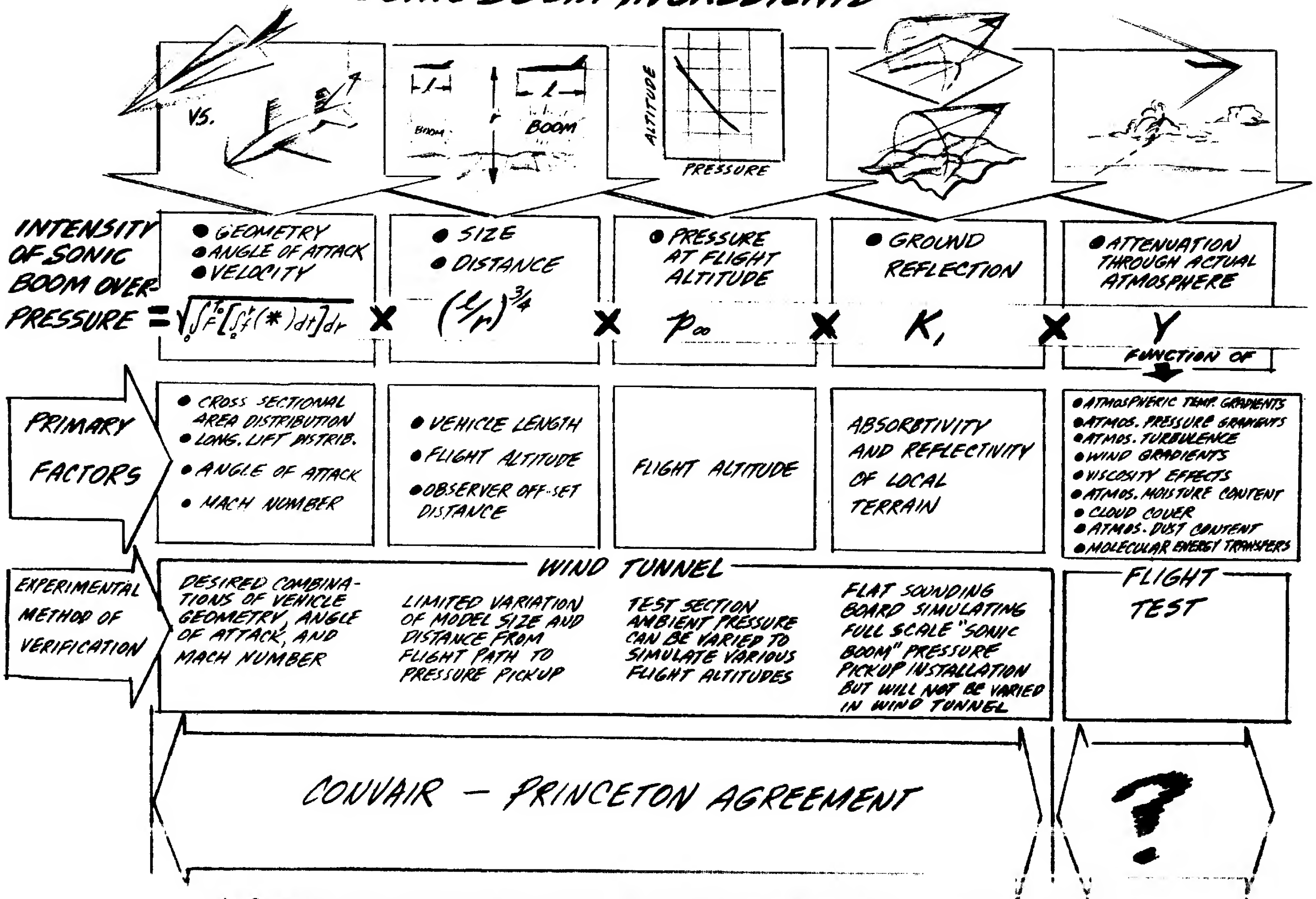
The first parameter is controlled by a combination of configuration geometry, angle of attack, and flight Mach number. For example, increases in configuration thickness ratio, angle of attack, or speed all tend to create a louder boom.

The second parameter is governed by the ratio of aircraft size (as typified by its length) and the distance from the aircraft flight path to the observer.

The pressure at the flight altitude is the third parameter. The boom overpressure is directly proportional to this pressure.

FIGURE 1

SONIC BOOM INGREDIENTS



* 2nd DERIVATIVE OF CROSSSECTIONAL AREA, LONGITUDINAL GRADIENT OF LIFT, & MACH NUMBER

The influence of the local terrain on the absorption and reflection of incoming acoustical waves is the fourth parameter. As an example, free air overpressures are doubled by the ground reflection from a flat surface such as a paved street or sidewalk.

Each of the above four parameters can be investigated by carefully planned wind tunnel experiments. And it is of interest that the Convair and Princeton boom analyses agree on the form of each of these parameters. The only disagreement between the two is over the fifth parameter, the attenuation of the boom through an actual atmosphere.

The inability to achieve controlled gradients in pressure, density, temperature, moisture and other properties precludes use of wind tunnels to explore the attenuation through real atmospheres. To achieve this, actual flight tests must be conducted wherein boom measurements are recorded from supersonic aircraft flying overhead. Two existing flight articles should yield the necessary data to determine actual attenuation. These are the B-58 bomber and the Bomarc missile.

Attention is called to the fact that F-101 and F-104 aircraft are commonly utilized by Convair, Fort Worth, as "chase planes" during B-58 flight test operations. Sonic boom recordings of these aircraft could be conveniently obtained over the same test range and under the same test conditions as the B-58. No cost

estimates are included herein for the addition of these two supersonic aircraft to the proposed B-58 sonic boom data program. However, it is pointed out that no additional instrumentation or manpower would be required in the flight test program. Additional model construction costs to be encountered in adding small models of these aircraft to the proposed wind tunnel program would be slight.

Convair believes that boom predictions for arbitrary vehicle geometries and flight conditions can be reliably determined from a coordinated wind tunnel and flight program of the B-58 and Bomarc. Analysis of these should yield a reliable method of extrapolating wind tunnel boom measurements of any particular supersonic vehicle to full scale operating conditions. The four steps of this program are:

1. Wind tunnel tests of the B-58 and Bomarc.
2. Flight tests of the B-58 and Bomarc.
3. Determination of generalized "scale" effects from analysis of the B-58 and Bomarc flight and model data.
4. Correlation of wind tunnel boom measurements of the particular vehicle by the results of Step 3 to attain full scale sonic boom predictions in the earth's atmosphere.

The proposed test programs and data analysis plan are presented in the following pages.

III. Wind Tunnel Tests

The configuration geometry, size, and Mach number as well as observer distance and pressure effects will be investigated in the N.A.S.A. Lewis Research Center 10 foot supersonic wind tunnels. Small, inexpensive models of the B-58 and Bomarc will be used to generate "sonic boom" overpressures which will be recorded at various stations throughout the test section as shown in Figure 2. Models will range from 2 to 12 inches in length.

The 10 foot supersonic wind tunnel at Lewis is capable of Mach numbers up to 3.6. "Boom" overpressures have not been measured in this tunnel but the needed instrumentation and test techniques have already been developed by the Langley Research Center and can be made available to Lewis.

A table summarizing the wind tunnel program is presented in Figure 3.

FIGURE 2

WIND TUNNEL TESTS

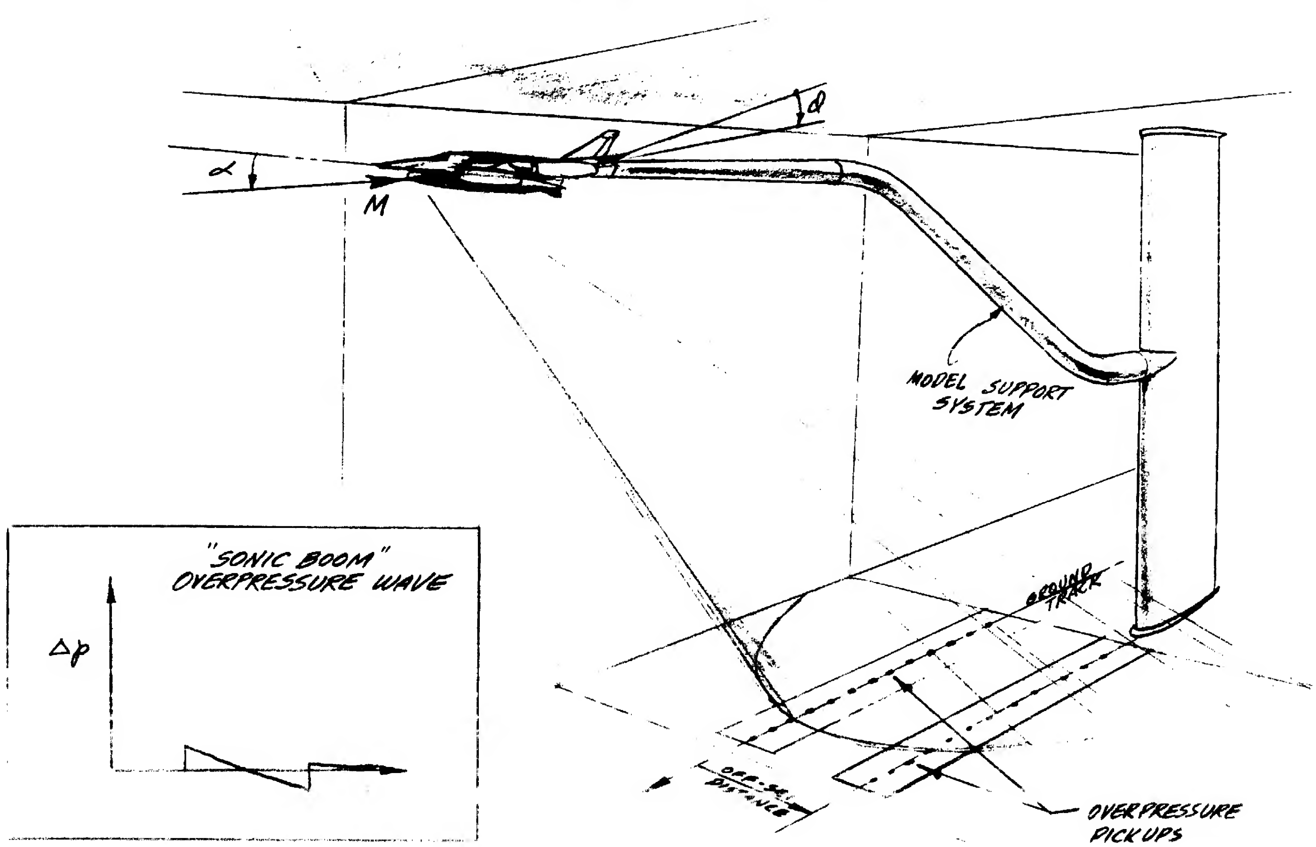


FIGURE 3

WIND TUNNEL PROGRAM

FACILITY	EFFECT TO BE INVESTIGATED	MODELS	TEST VARIABLES
LEWIS RESEARCH CENTER 10 FT. SUPERSONIC WIND TUNNEL	SIZE - DISTANCE	B-58 BOMARC	• MODEL SIZE • OVERPRESSURE PICK-UP LOCATION
	CROSSECTIONAL AREA DISTRIBUTION	B-58 BOMARC	MODEL CONFIG.
	ANGLE OF ATTACK AND VELOCITY	B-58 BOMARC	• MODEL ANGLE OF ATTACK, • MACH NUMBER
	FLIGHT ALTITUDE	B-58	TEST SECTION AMBIENT PRESSURE

IV. Flight Test Program

Selection of the B-58 and Bomarc vehicles for the proposed flight test program has been dictated by their abilities to generate controlled data at high Mach numbers and extreme altitudes. Data will be gathered in performance realms assigned to each vehicle as follows:

Vehicle	Mach Number	Altitude
B-58	1.5 - 2	40-60,000 ft.
Bomarc	2.6 or higher	60-70,000 ft.

This data will be recorded on several different test dates in order to obtain a reasonable sampling of the effects of atmospheric irregularities. Temperature and wind distributions, cloud covers, and atmospheric moisture content existing at the time of each test run will be recorded. This information will be needed to guide the data reduction, analysis, and correlation required to define atmospheric attenuation effects. Vehicle Mach number, altitude, and offset distance must be recorded on every test run.

Convair and N.A.S.A. personnel have arrived at common decisions relating to instrumentation techniques. Parallel but independent experience shows that "sonic boom" overpressures should be recorded with instrumentation systems having both high and low frequency response capabilities. The proposed technique calls for simultaneous use of three (3) different type pressure transducers at each data gathering station. The C.E.C. Type 4-340 Sound Pressure Level

Pickup has been selected due to its wide-band frequency response and low unit cost. An Altec-Lansing Model 21 BR(150 db) Condenser Microphone and a Statham Model P97TC-0.05D-350 Pressure Transducer are called for to permit correlation with earlier data recorded with these instruments. All this equipment is readily available commercially. Additional instrumentation details are covered on later pages.

All B-58 data will be obtained by Convair as a part of its present flight test program. Three acoustical detection stations will be required to properly cover the boom swath as shown in Figure 4. Lined up perpendicular to the flight corridor they will insure adequate coverage in spite of inevitable deviations from the prescribed flight path. The width of this path is revealed in Figure 5 which shows the path and resultant reported complaints arising from the booms of a recent B-58 test flight. The width of this boom swath is indicated to be about 100 nautical miles, roughly twice the theoretically predicted value. A reported seismograph reading at the time of this "sonic boom" incident incited local news reports of an unprecedented "Amarillo Earthquake."

During the proposed B-58 test period it would be possible to achieve additional useful data associated with sonic booms. For example, the nose boom of a chase airplane used in the testing could be pressure instrumented to directly verify the free air pressures surrounding the B-58 as measured in the wind tunnel. These data would be of value to USAF in determining the effect of a boom from one aircraft striking another flying in the vicinity.

FIGURE 4
B-58 FLIGHT TEST PROGRAM

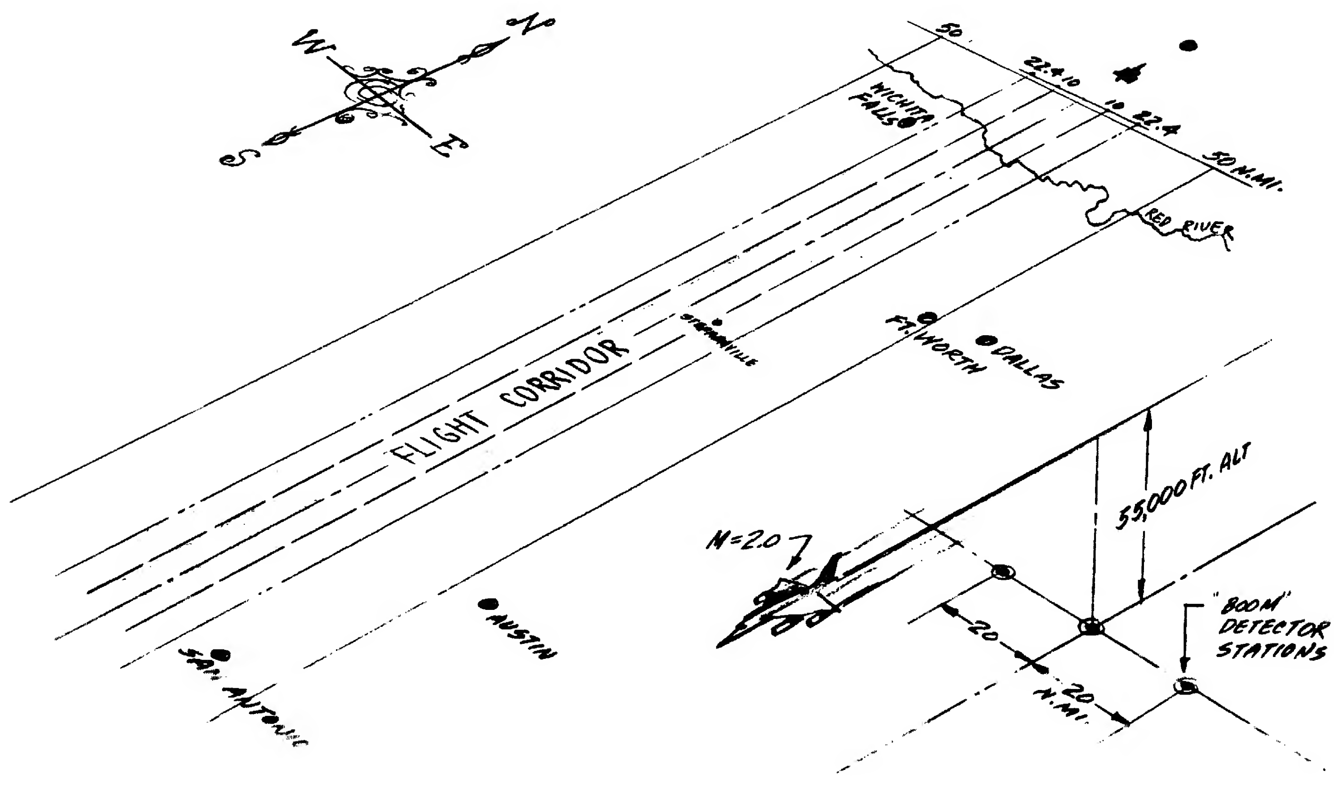
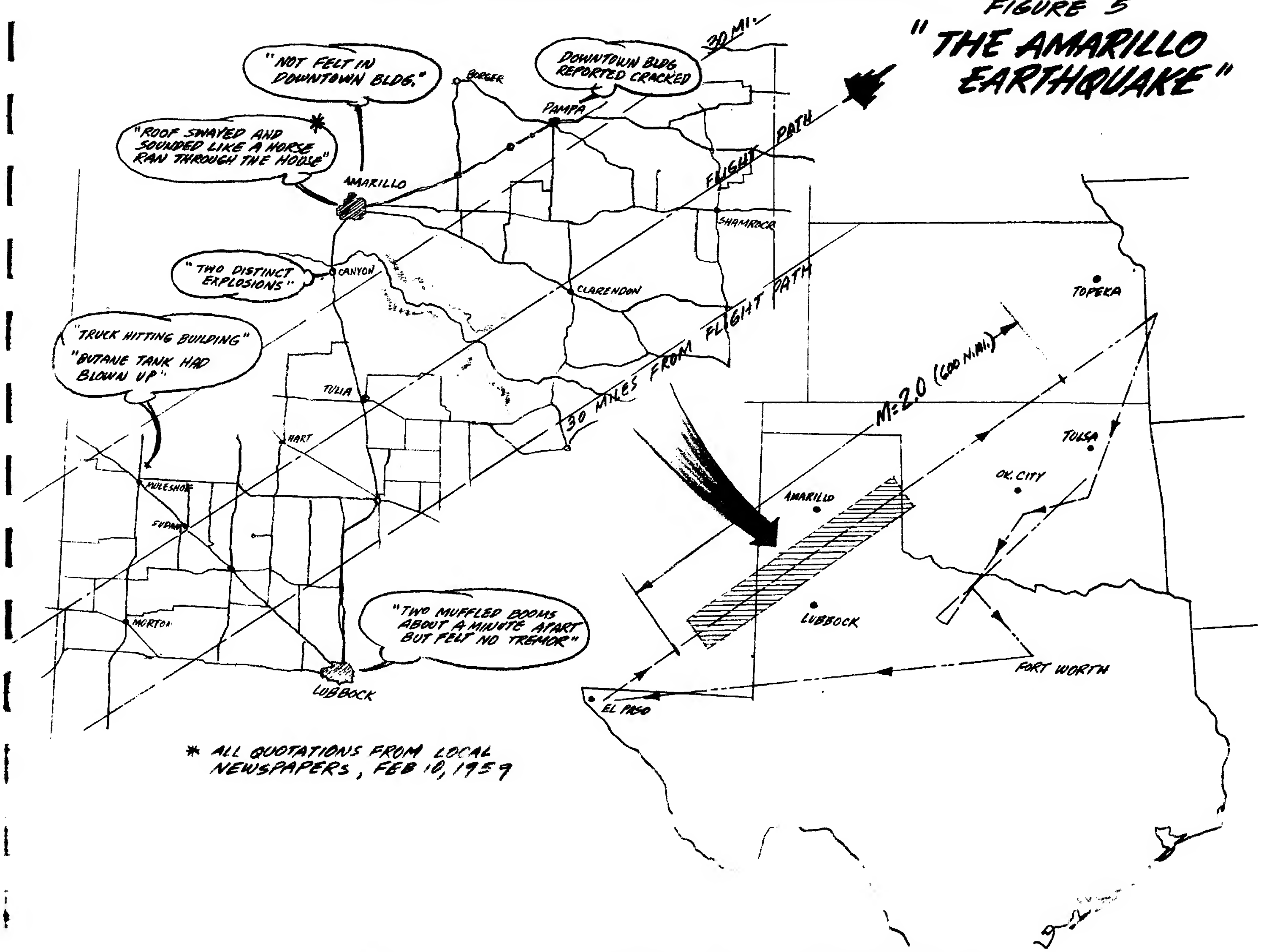


FIGURE 5

"THE AMARILLO EARTHQUAKE"



* ALL QUOTATIONS FROM LOCAL NEWSPAPERS, FEB 10, 1959

The Bomarc boom data may be gathered at two Florida sites in the near future. During the months of March and April, the Boeing Airplane Company will conduct two more firings at Cape Canaveral suitable for boom measurements.

Starting "soon" (presumably in March) USAF is to commence training firings of Bomarc from an Eglin Field installation. The rate of firing is understood to be one per week. This series of firings should prove quite beneficial in generating valuable boom data.

In addition, B-58 aircraft #8 is currently scheduled to be based at Eglin Field for fire control testing from mid-March through December 1959. This aircraft might conveniently be flown over the sonic boom data recording range proposed for Bomarc in order to gain "sonic boom" data for the B-58 and Bomarc under the same atmospheric conditions. Such flights would insure correlation of Bomarc flight test data with that recorded in the B-58 flight test program at Fort Worth.

As sketched in Figure 6, five small craft (patrol or crash boats) would be positioned about 40 miles to sea in a line normal to the flight path. Aboard each boat would be a complete boom detection package.

Figure 7 gives a comparison of predicted and recorded boom levels of the B-58. Predicted ranges of boom levels anticipated from larger and smaller supersonic vehicles such as a typical 500,000 lb. bomber and a typical 15,000 lb. missile are also illustrated in this figure.

FIGURE 6

BOMARC FLIGHT TEST PROGRAM

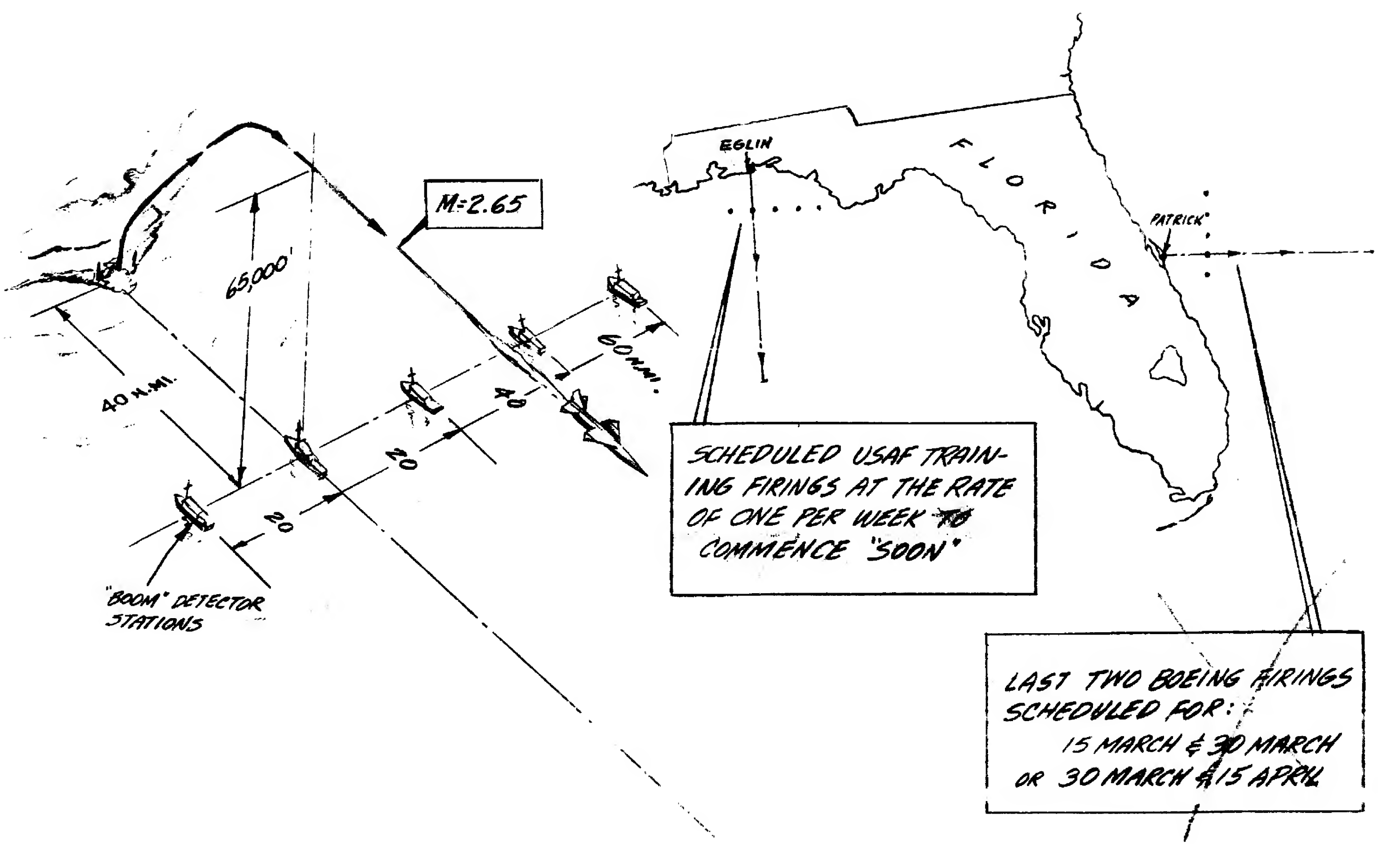
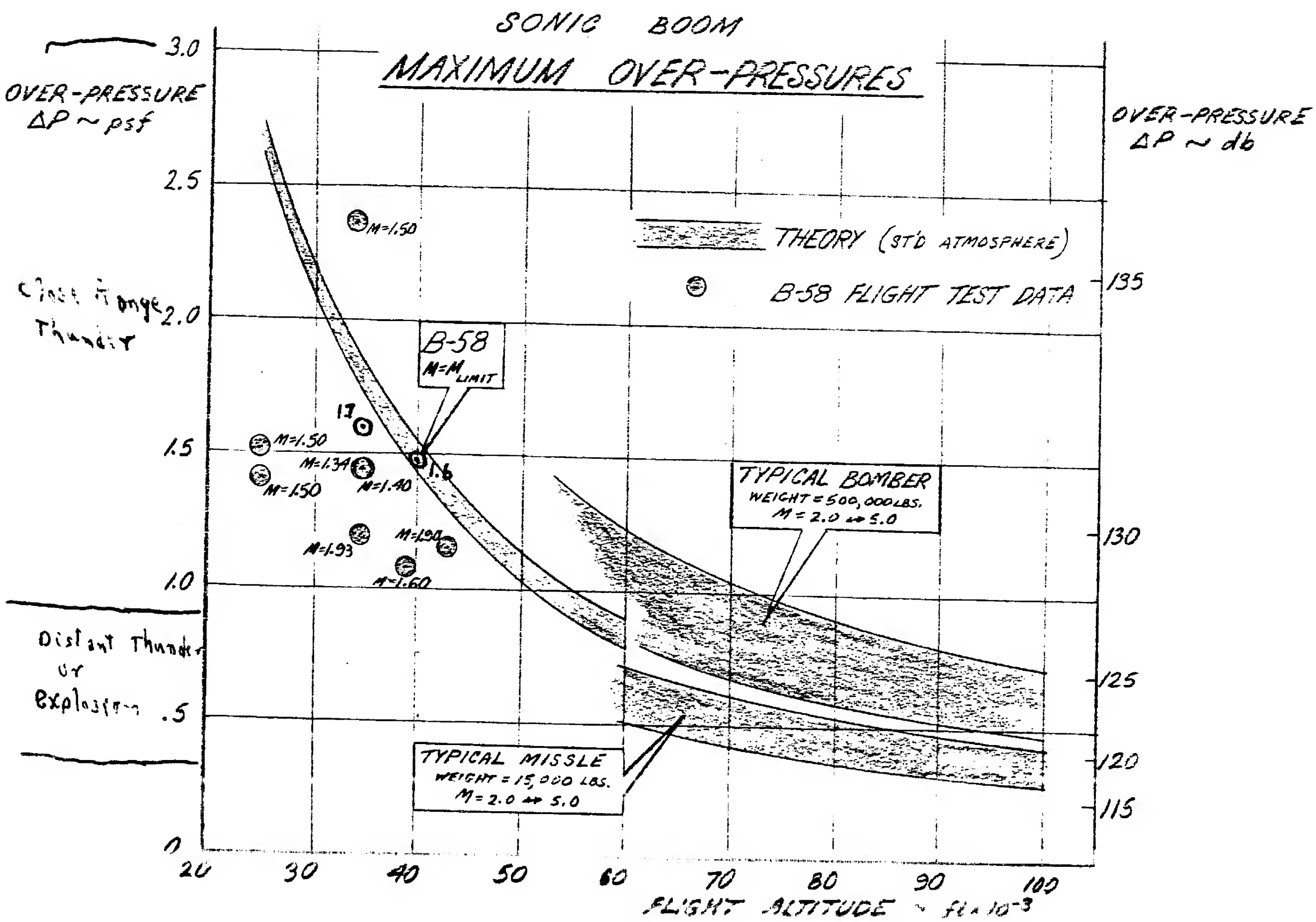


FIGURE 7


SONIC BOOM
MAXIMUM OVER-PRESSURES
OVER-PRESSURE ΔP ~ psf
OVER-PRESSURE ΔP ~ db
3.0
2.5
2.0
1.5
1.0
.5
0
135
130
125
120
115
Close Range Thunder
Distant Thunder or Explosion
THEORY (STD ATMOSPHERE)
B-58 FLIGHT TEST DATA
B-58
M=M LIMIT
TYPICAL BOMBER
WEIGHT = 500,000 LBS.
M = 2.0 → 5.0
TYPICAL MISSLE
WEIGHT = 15,000 LBS.
M = 2.0 → 5.0
M=1.50
M=1.34
M=1.40
M=1.93
M=1.90
M=1.60
1.6
17
20
30
40
50
60
70
80
90
100
FLIGHT ALTITUDE ~ ft x 10^-3

V. Method of Data Analysis

Convair's analytical approach to the "sonic boom" prediction problem is essentially the same as was adopted in the Princeton Report. The effects of configuration geometry and size and Mach number are identical in both analyses. However, differences in the prediction of "sonic boom" intensity result from differences in the atmospheric attenuation effects assumed by each organization.

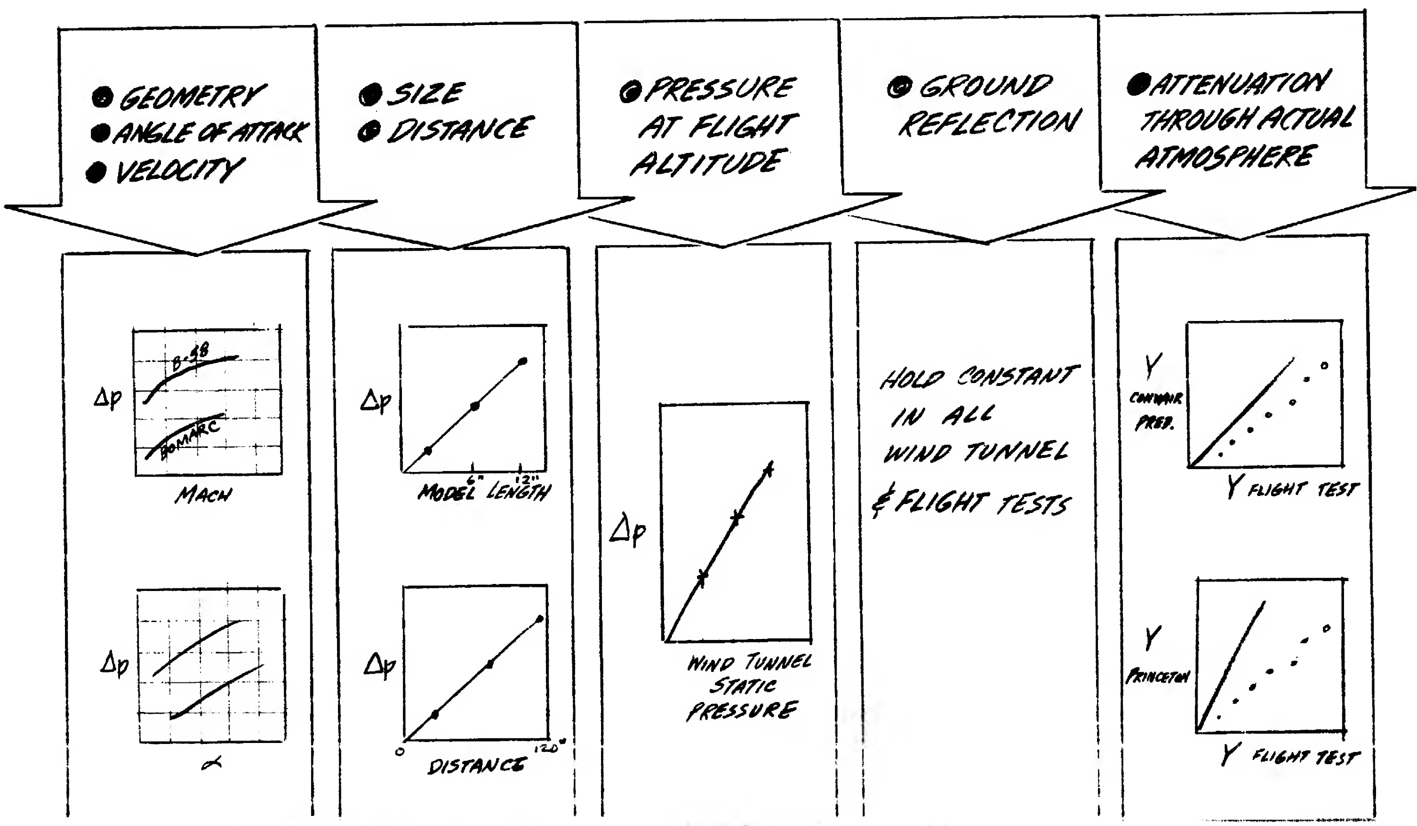
Wind tunnel test results will be summarized and presented in the form of the basic parameters common to both Convair's and Princeton's analyses. Isolated effects of configuration geometry and size and Mach number will be clearly defined. These data summaries will provide reliable predictions of initial "sonic boom" intensities, prior to any atmospheric attenuation or refraction, for the B-58 and Bomarc.

An evaluation of atmospheric attenuation and refraction effects will be derived from direct comparison of wind tunnel and flight test data. The flight test data will be analyzed to determine the effects of non-standard temperature gradients, wind gradients, cloud cover, and atmospheric moisture content. A knowledge of the effects of these phenomena upon the properties of sonic booms is prerequisite to accurate predictions of booms propagating through real atmospheres.

Figure 8 illustrates the proposed method of data analysis. The immediate value of this proposed data analysis in planning military operations with present day and forthcoming supersonic vehicles is obvious.

FIGURE 8

DATA ANALYSIS

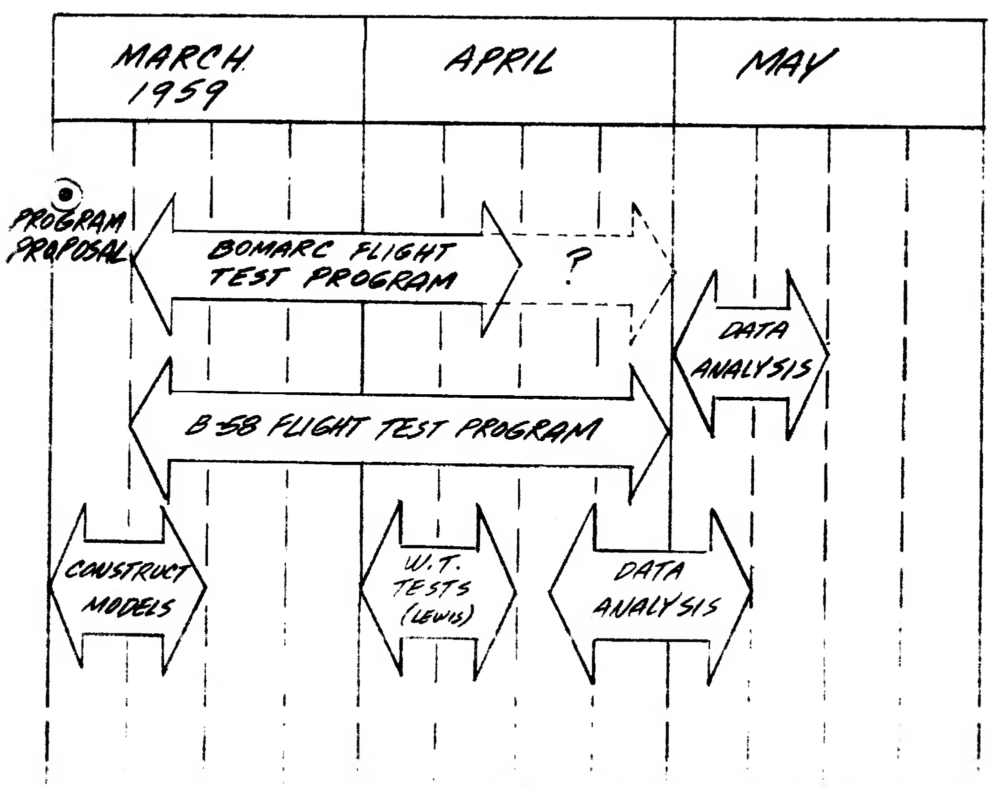


VI. Tentative Schedule

The tentative schedule, illustrated in Figure 9, has been set in accordance with the immediate needs for the proposed data. Implementation of the flight test program, in particular, should be initiated at the earliest possible date. Previous experience gained by Convair personnel in gathering "sonic boom" data indicates that considerable time may be required to accrue the proposed flight test data. Favorable combinations of atmospheric conditions and scheduled flight tests may be anticipated on a statistical basis only. It is proposed that the wind tunnel tests be conducted at the earliest possible date also. Thus adequate time may be allowed to analyze results and properly evaluate these by a 16 May target date.

FIGURE 9

PROPOSED SCHEDULE



VII. Estimated Costs

This section has been prepared in an effort to provide the data required to make an overall cost estimate for the proposed program. Expense items which can be estimated by Convair amount to a total of $63,198. This total is herein broken into three pertinent areas for further analysis and approval.

A. Wind Tunnel Program

The following cost estimates for the proposed Wind Tunnel Program assume that test time will be made available, free of charge, in the aforementioned NASA Lewis 10 foot supersonic wind tunnel. It is further assumed that this test program will be conducted in conjunction with an NASA "sonic boom" research program. Experienced personnel and proven instrumentation accruing from previous "sonic boom" research at NASA Langley will be utilized in conducting the program currently proposed. Additional costs are as follows:

Model Construction Costs	$ 8,800
Transportation Expenses	1,030
Per Diem Expenses	500
TOTAL	$10,330

B. **B-58 Flight Test Program**

The following cost estimates assume that all B-58 "sonic boom" data gathering will be conducted by Convair, Fort Worth personnel in the local vicinity of that installation. It is estimated that twelve (12) local field trips will be required to obtain the needed satisfactory test runs which will be recorded at three separate stations. Data gathering will be phased in with the current B-58 Flight Test Program. Costs to be incurred in equipping and operating three (3) data stations are as follows:

Manpower (1140 hrs. @ $6.45/hr. avg. cost)	$ 7,353
*Instrumentation to be Purchased	12,490

2 C.E.C. Transducers @ $175 each
2 Statham Transducers @ $300 each
2 Altec-Lansing Transducers @ $300 each
2 Oscillator Power Supplies for Microphones at $200 each
6 Driver Amplifier and Pre-Amplifier Units at $610 each
2 Recording Oscillographs @ $2800 each
8 Galvanometers @ $160 each

Materials required	300
TOTAL	$ 21,143

*Note: This estimate assumes that one (1) of three (3) data stations [illegible] equipped with instrumentation currently available at this installation. It may be possible to reduce this instrumentation purchase expense [illegible]

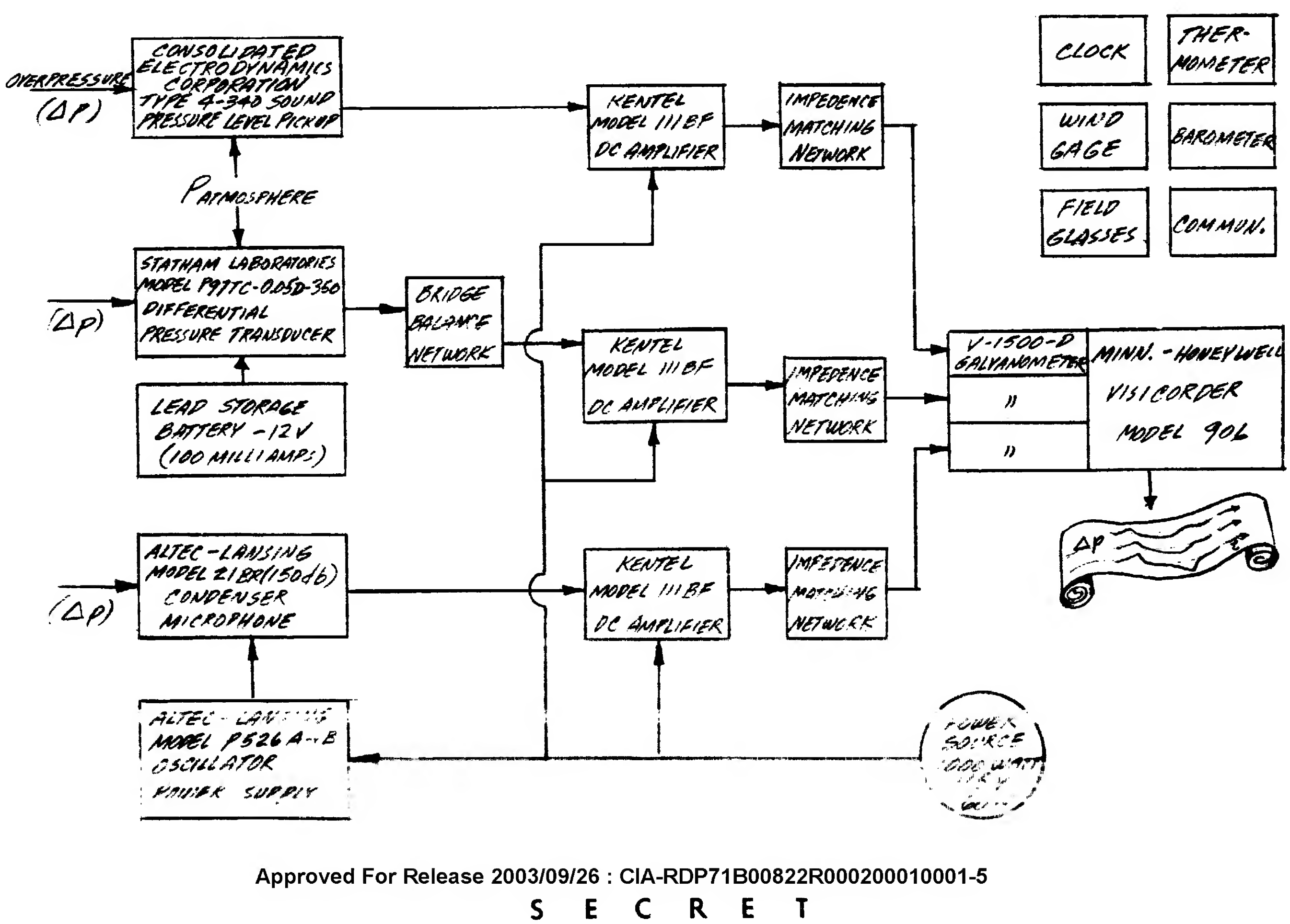
C. Bomarc Flight Test Program

It is assumed that these data will be gathered in the vicinity of Cape Canaveral, Florida. Basic support equipment and personnel to man five (5) acoustical recording stations are assumed to be furnished by the data gathering agency. Implementation of this program and utilization of any Convair personnel have not been considered. Costs for equipping the stations with recommended pressure sensing instrumentation illustrated in Figure 10 are as follows:

*Instrumentation Costs (per station)	$ 6,245
Material Costs (per station)	100
Total per station	$ 6,345
Total for five (5) stations	$ 31,725

*Note: Proposed instrumentation of each data station and appropriate unit costs are detailed in the following section. Reference to that section will enable the above costs to be reduced in accordance with on-site available equipment inventories. Attention is called to the fact that the N.A.S.A. Langley Instrument Pool inventory includes much of the needed equipment. It is believed that two acoustical detection stations could be instrumented with equipment currently available there. Equivalent instrumentation for five stations could be implemented by drawing equipment from other projects there and appropriating only six pickup transducers not in their current inventory.

FIGURE 10
FLIGHT TEST DATA INSTRUMENTATION



VIII. Suggested Instrumentation

It is proposed that each data station be furnished three (3) particular pressure transducers and appropriate amplifying and recording equipment as follows:

1	Consolidated Electrodynamics Corporation Type 4-430 Sound Pressure Level Pickup	$ 175
1	Statham Laboratories Model P97TC-0.05D-350 Differential Pressure Transducer	300
1	Altec-Lansing Model 21 BR (150 db) Condenser Microphone	300
3	Driver Amplifiers and Pre-Amplifiers @ $610 per channel (any models capable of 4 to 500 cps flat frequency response)	1,830
1	Oscillator Power Supply for Microphone	200
1	8-channel Honeywell Visicorder	2,800
4	Galvanometers for Recorder @ $160 each (0 to 500 cps flat frequency response)	640
	Total per station	$ 6,245

SHOCK WAVE EFFECTS DUE TO HIGH ALTITUDE SUPERSONIC FLIGHT

THIS SUBJECT WAS DISCUSSED WITH MR. FRED DAUM, AERONAUTICAL RESEARCH LABORATORY AND DR. H. O. PARRACK, COORDINATOR AIR FORCE NOISE & VIBRATION PROGRAM, WRIGHT FIELD.

A BRIEF RECAP: THERE HAS BEEN NO HIGH ALTITUDE SUPERSONIC FLIGHT TEST FOR THIS PURPOSE ABOVE 50,000 FEET ALTITUDE. PHASE II OF "LITTLE BOOM" RECENTLY CONDUCTED, WAS A HIGH ALTITUDE (BELOW 50,000 FEET) SUPERSONIC TEST AND THE REPORT IS BEING PREPARED BY NASA AT LANGLEY FIELD, VA. AND SHOULD BE RELEASED SOON.

TWO EARLY REPORTS BY MR. FRED DAUM ARE ATTACHED. A NATO REPORT DATED SEPTEMBER 1959 IS AVAILABLE. SEE ATTACHED SUMMARY.

AN EXTENSIVE TEST PROGRAM IS BEING PLANNED AS A PRELIMINARY TO DEVELOPMENT PLANS FOR A SUPERSONIC COMMERCIAL AIR TRANSPORT. THIS TEST PROGRAM PLANNING INVOLVES FAA, NASA AND USAF. DR. PARRACK IS CONCERNED WITH THE USAF PARTICIPATION.

Capt Chapman

COVER SHEET FOR SECRET DOCUMENTS

SECRET

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SECRET

"The Theory and the Problems of the Sonic Boom"

A Presentation to the Air Force Committee to
Review Present Long Range Air Base Problems

Presented by Mr. Fred L. Daum
Aeronautical Research Laboratory
Wright Air Development Center
Air Research and Development Command

6 October 1954

Carpenter Litho & Prtg. Co., Springfield, O.
500 - 17 January 1955

"The Theory and the Problems of the Sonic Boom"

The sonic boom phenomenon is not something new but has actually been observed in various forms for many years. For example, a small sonic boom is created by the cracking of a whip, the end of which actually accelerates to sonic speeds and generates shock waves which when passing the human ear are heard as cracks. Also, in ballistics firings it was noted years ago that supersonic projectiles made a cracking noise as they passed by. In both of these cases it was recognized that the noise was the result of shock waves generated during supersonic motion.

Apparently no particular thought was given by the aerodynamicist to the fact that shock waves generated by aircraft flying at supersonic speeds would, in addition to increasing the aircraft drag, also be heard by observers as explosion-like sounds. Thus, the first occurrence of the sonic boom resulting from diving the early F-86 airplanes to slightly supersonic speeds came as a surprise as mysterious unexplained explosions began to appear across the country. Finally, in about March 1950, several test dives at Wright-Patterson Air Force Base showed definitely that the diving F-86 did create the booms heard so mysteriously. Then a number of theories regarding the phenomenon appeared, all trying apparently to account for the varying number of booms which were heard on different occasions.

Recently aircraft performance has been increased to the point where low supersonic speeds can now be attained in straight and level flight. It also appears reasonable that, in the not too distant future, speeds up to several

times the speed of sound will be reached. The knowledge that property damage and public panic have resulted in the past with aircraft that barely exceeded the speed of sound, and then for only short periods of time and at high altitudes, quite naturally leads to the question of, "What will be the effects of these booms from airplanes flying at much greater speeds and at lower altitudes?", and this we will discuss later.

Before going further, let's answer the question, "What is a sonic boom?" The answer is that a sonic boom is a sound which is heard when a shock wave, which is a pressure wave, is created by a body moving at supersonic speed and reaches the ears of an observer. Sound waves are pressure waves, and a single strong pressure wave is therefore heard as a boom. The number of booms which are heard is the number of shock waves which reach the observer.

Now let's take a minute to see, "What are shock waves?" and, "How are shock waves generated?" We cannot see the shock waves in air, but an analogy which we all have seen arises from the case of a boat moving through water; the water piles up immediately in front of the boat and water waves trail rearward at an angle from the bow and stern. Further, if you were sitting in a small boat and observed the waves being made by another boat as it passed, you also noticed that your boat rocked as the waves passed. Also, there probably have been times when the passing boat was large or passed near by which made large waves and you became quite concerned about your boat upsetting! A similar thing happens in air in supersonic flight where the waves are differences in air pressure and correspond to the difference in water level in the water-wave analogy. An item worth noting here, is the fact that in the propagation of the

water wave there is no continuous flow of water in the wave direction but rather it is a temporary distortion of the surface which moves along. Also, for weak pressure waves in air, there is no continuous movement, or flow, in the direction of the wave but only a small movement where the air jumps ahead, kicks the air in front of it in the seat of the pants and then quickly jumps back. Meanwhile, the kicked air passes the kick on and in this manner the pressure wave travels through the air at the speed of sound, if the disturbance is weak.

It is these pressure disturbances which travel ahead of a body in subsonic motion and warn the air ahead that the body is coming so that the air begins to move aside for the body to pass before the body gets there. Now if the body moves at supersonic speed, these disturbances are not moving as fast as the body and so there is no way of warning the air ahead that the body is coming. The air in this case does not feel the presence of the body until it actually arrives and then very quick action is required for the air to move aside.

In order to clarify this point let's look at the following chart (Figure 1) which illustrates a simplified version of the propagation of pressure disturbances. Here, a stationary source of pressure disturbances is shown where the pressure waves propagate radially away from the source at the speed of sound.

Next, consider that the generating source is moving at a subsonic speed. Each disturbance generated is still radiated in all directions from the point where it was created. It is seen that there is a tendency for the disturbances to become more closely spaced in the direction of

motion because the instantaneous centers of the disturbance rings are no longer superimposed, as in the stationary source case, but are distributed along the path of the motion.

Now consider that the source generating the pressure disturbances is moving faster than the speed of sound. This means that the generating source actually moves outside of the disturbance ring just generated, as shown on the chart. It is also shown that the disturbance rings all become tangent to sloping lines drawn rearward from the source. These lines are regions where the individual disturbances reinforce each other and build up large disturbances. These large disturbances are shock waves. Looking at the geometry of the figure it is seen that the wave must be moving away from the path of motion in a direction normal to the wave front and at the velocity of sound. This geometry provides a clear picture of the term "Mach number", which is the ratio of the disturbance velocity to the sound velocity and the term "Mach angle", which is the angle the shock wave makes with the path of motion. This was not intended as a rigorous explanation of the formation of shock waves and a number of slight variations do appear but this should suffice for the present purpose.

At this time it is appropriate to view a short film which has been prepared by North American Aviation, Inc. on the subject of the sonic boom. This film deals with a shallow water wave anology and further demonstrates some of the points already discussed. The airplane model shown does not touch the water but sits upon a support having a double wedge or diamond shaped section which is in the water; it is the support that makes the waves.

The first scene shows an object being dropped into a quiet water surface and the disturbance waves are seen propagating radially away from the disturbance source at a speed which corresponds to the speed of sound in air. Next the steady state motion case is shown; as the body moves through the water at a steady speed the head and tail waves are seen trailing from the body. Next the waves are shown to form as the body is accelerated. The effects of slowing from supersonic to subsonic speeds are next shown; this scene illustrates the transient case where the waves propagate ahead of the body at about the sound velocity as the body slows down. This last case will be discussed further, later on.

The shock wave concept is now the generally accepted theory of the sonic boom and some of the early ideas that a sharp pull-out is required or that the boom results from an accumulation of sound have been abandoned. The important variable, in relation to the sonic boom, is the magnitude of the pressures associated with shock waves, since these pressures determine the overally loudness of the boom, the possibility of damage to property, and the likelihood of adverse physiological effects on the human being. Therefore, we will discuss the pressure characteristics of shock waves.

Our discussion of the pressure characteristics of shock waves will center around the changes in pressure which occur during the passage of a shock wave Also we will consider the pressure changes in terms of the units, "pounds per square foot". In order to get a feeling of what these "pounds per square foot" numbers mean, as far as the hearing sensation in the ear is concerned, the following chart (Figure 2) has been prepared. We see here the ear is apparently intended to operate over the sound pressure level range of about 0.0005 to 2.0 pounds per square foot, where the sound heard at the 2.0 pounds per square foot

pressure level is very loud. We see that at 4.0 pounds per square foot the sound becomes painful to the listener and at sound pressures of about 40 pounds per square foot and above, human ear drums begin to rupture.

Let's switch our discussion here to the shock waves which are associated with a typical airplane flying at supersonic speed. Looking at the next chart (Figure 3) two cases are shown. One is for straight and level flight which gives a steady state condition; the other case is for a diving flight with a pull-out and which gives transient conditions and the booms which most of us have experienced so far. For the steady state condition, the airplane and the shock waves move through the air as a system. The wave pattern near the airplane has been simplified on the chart and only the primary waves are shown; in general these are the main waves with which we are concerned. If the wave system shown extends downward from the airplane a sufficient distance to reach the ground, then the waves travel along the ground at the same speed as the airplane. At each point on the ground which the waves cross, two booms will be heard as the bow and tail waves pass. The airplane in this case leaves a continuous trail of booms along the ground under the airplane. The pressures on the ground at any point which the waves cross would vary about as shown; that is, there would be a strong compression (the first boom) followed by a gradual decompression to a pressure below atmospheric, and finally a strong recompression to atmospheric pressure (the second boom).

The diving airplane presents a different picture in that the airplane pulls out of the supersonic dive and slows to a subsonic speed while the shock waves, which are propagating at velocities generally

slightly greater than, but near, the sound speed, pass the airplane and continue on their own, eventually striking the ground and creating booms.

Theoretical Considerations of Shock Wave Pressures

The theoretical aspects of the transient phenomenon are quite complicated because of the varying Mach number and the usually curved flight path and have not yet been very thoroughly investigated. The steady state case has been theoretically analyzed although many simplifying assumptions were made in the analysis. The following chart has been prepared (Figure 4) to show the theoretical variation, with distance from the body, of the jump in pressure across the bow shock wave. The curves were obtained through application of the results of an analysis of the behaviour of shock waves produced by bodies of revolution performed by the British Mathematician Mr. G. B. Whitham. For this, and the following theoretical data which are presented, it is assumed that the body moves through still air of uniform pressure and temperature. The theory further applies to a parabolic body of revolution with the length of a typical airplane. This plot is aimed at demonstrating the effects of increasing Mach number and curves are shown for Mach numbers 1.04, 2, and 4, since it is likely that within the next 10 years level flight up to perhaps Mach 4 may be achieved. As seen from this plot, the effects of Mach number are small especially at distances greater than 5,000 feet. Here, at Mach 1.04 the peak pressures 5,000 feet away from a typical airplane amount to only about 4 pounds per square foot and increasing the Mach number to as high as 4 only raises the peak pressure to about 7 pounds per square foot. It was shown earlier that this 4 pounds per square foot will result in a

painful sensation in the human ear. At greater distances from the airplane the waves are shown to grow very weak. It will be noted, however, that in going closer than 5,000 feet the peak pressures observed rapidly increase and at a Mach number of 1.04 and 200 feet away the pressure expected would be about 40 pounds per square foot. Actually, at this speed, the pressure jump immediately in front of the nose of the airplane amounts to 195 pounds per square foot. In order to more clearly illustrate the effect of Mach number on the bow wave pressure jump at close distances the following chart (Figure 5) has been prepared and shows, for the size of a typical present day fighter type airplane, the pressures that might be experienced if the airplane passed over head at only 200 feet. This clearly indicates that the Mach number effects are not strong, since doubling the Mach number of 1.3 gives only about a 30 percent increase in pressure jump.

Now let's consider theoretically what are the effects of varying the body size, again, for a parabolic body of revolution. This next chart (Figure 6) shows the variation of pressure jump across the bow wave with distance from the body, for a flight Mach number of 1.04 and for bodies of various slenderness ratios. As would be expected, the larger body diameter results in greater pressure jumps. If extended down to the very front of the body these 4 curves would each have to show the same pressure of about 200 pounds per square foot. Here, we have held the body length constant and varied the diameter. We see that the pressure jump is then directly proportional to the body diameter-to-length ratio. If we keep the body shape fixed and merely enlarge the same body, the pressure jump at any given distance from the body is then proportional to the 3/4 power of the body length; the pressure does not increase quite

as fast as the body size. This theory is not valid for distances very close to the body. The curve for δ = .15 is fairly representative of todays airplanes, where δ is the ratio of maximum cross section diameter to the body length.

We have assumed so far that the body generating the shock waves is flying in air at sea level pressure. For the purpose of illustrating the effects of altitude, or decreasing static pressure, on the magnitude of the pressure jump across the bow wave, the following chart (Figure 7) has been prepared. This chart clearly shows that with increasing altitude the pressure jump becomes smaller. The reason for this is that the body size, flight Mach number, and the distance from the body determine what the pressure ratio across the shock wave will be; that is, the ratio of pressure behind the shock to that in front of the shock. Thus, if the pressure in front of the shock is decreased then the pressure jump will decrease in order that the pressure ratio remains the same. Then, at 20,000 feet altitude where roughly only one half an atmosphere pressure exists, the pressure jump is only half of that which would be obtained at sea level.

Also in the case of flight through the atmosphere there is a decrease in temperature with increasing altitude which roughly amounts to 4.5 degrees Fahrenheit per thousand feet, or converseley, in coming down from altitude the temperature increases. As was shown earlier the speed at which the shock waves propagate, at appreciable distance from the airplane, is equal approximately to the speed of sound. The wave speed actually depends on the speed of sound and the speed of sound, in turn, depends upon the

absolute temperature of the air. Therefore, in the normal atmosphere, as the shock waves travel down to lower altitudes the bottom end of the wave begins to travel faster, which makes the wave curve forward. For an inverted temperature gradient the shock wave would curve rearward. Because of this temperature difference in the atmosphere it is possible to fly at low supersonic Mach numbers at altitudes where the velocity of sound is low and where shock waves are formed and are pulled along with the airplane, but, with respect to the sound velocity at the ground level the airplane is subsonic and the shock waves therefore cannot exist at the ground level. This means then that it is theoretically impossible for an airplane to fly in level flight at a speed less than 760 miles per hour (or a speed equivalent to the speed of sound at the ground) and have the waves which are being pulled along reach the ground, the reason being that this would result in an impossible case of having shock waves moving at subsonic speed. The following chart (Figure 8) has been prepared to illustrate the effects of atmospheric temperature on the steady state shock wave. With the normal temperature variation with altitude the wave is shown to run forward and stop above the ground at a position where the local sound velocity is equal to the airspeed of the airplane. The inverted temperature gradient causes the wave to sweep back at a greater angle, as shown.

The distribution of wind through the atmosphere also effects the shape of the shock wave front. The case of a wind increasing with altitude is shown on this chart (Figure 9). The airplane is shown flying into a headwind and at a given airspeed. With respect to the

air at lower altitudes the airspeed is lower since the head wind is less so that in moving down the shock wave the Mach number decreases. The effect is that the wave steepens at the lower altitudes since it must adjust to the decreasing Mach number. It must be remembered here that the shock waves and the airplane are all moving through the air at the same absolute speed as a system. Also shown on this chart (Figure 9) is the effect of the same wind gradient but with the airplane flying down wind. The expected effect in this case would be to bend the wave rearward, as shown. It should also be mentioned here that there is to be expected an effect of atmospheric pressure on the pressure jump across a shock wave as the wave goes to lower altitudes. The exact effects are not presently known but this situation is presently being analyzed and studied in detail along with the other atmospheric effects.

How Does Theory Compare with Experiment?

Let's take a look to see how theory and experiment compare. Only a limited amount of experimental data have been obtained to date. Some data were obtained from the transient case of the diving F-86 airplane and the next chart (Figure 10) shows a typical time history of the ground pressure for this case. It has been learned that the first N-shaped pressure wave is from the passing of the head and tail shock waves and the second weaker wave is the result of a transonic wave which begins building up slightly before the airplane reaches the velocity of sound and which drops away from the airplane to the rear after the airplane goes through Mach one. This second set of waves will not be discussed any further because it has always been observed to be weak relative to the first wave. Little more

than the magnitude of the ground pressures was learned from the early tests.

With the construction of several Air Force airplanes capable of supersonic straight and level flight, the WADC experimental efforts were extended. The data from this recent study are not yet completely reduced but the available results will be presented.

The following chart (Figure 11) presents the results of several close fly-bys at Mach 1.04 and compares these points with Whitham's theory. The experimental data were obtained during tests on a 10,000 foot high mountain peak but the results have been reduced to equivalent sea level pressure. The ratio of the effective maximum airplane body diameter to airplane length was about 0.17 for which this theoretical curve was computed. It is interesting to note that at 200 feet altitude and at Mach 1.04 a pressure jump of 28 pounds per square foot was measured which would have been about 36 pounds per square foot if the pass had been made at sea level. Also, an important point is that fairly good agreement exists between the theory and the experiment indicating that shock wave pressures can be predicted with reasonable accuracy using the available simplified theory.

An interesting plot is shown on the following chart (Figure 12). The solid line represents the theoretical maximum altitude at which straight and level supersonic flight can be accomplished and have the shock waves just reach the ground. This accounts only for the standard atmospheric temperature effects and neglects wind and pressure influences. The limiting line is shown to rise vertically from 35,000 feet altitude because the temperature of the standard atmosphere is essentially constant between 35,000 and 100,000 feet altitude. Experimental verification

of this curve has been loosely established. Also, at an altitude of about 30,000 it presently appears from experiment that the curve will bend over something like that shown by the dashed line. This plot indicates that supersonic straight and level flight is possible without creating booms as long as the flight conditions fall in the region above the solid and dashed lines. It appears possible that the curve will level off between 35,000 and 40,000 feet altitude. This altitude range then, represents for straight and level flight the limiting distance which these waves will propagate down through the standard atmosphere and remain audible, considering only the effects of atmospheric temperature.

Next we will consider the ground area covered by these waves. From observations made by persons spread along a line, first parallel to the flight path and then perpendicular to the flight path, the following chart (Figure 13) has been prepared. The lateral spread measured was not necessarily a maximum. This is also for the case of the airplane diving to 20,000 feet altitude at supersonic speed. The chart shows that for both directions the sound was heard over a ground distance of between 15 and 35 miles. This is only qualitative data but does serve as a guide regarding the range of the waves and would seem to discount some of the random reports which have been made regarding the extreme range supposedly covered by the waves.

Possible Damage From Sonic Boom

The question of, "How much damage can be done by the boom?" is difficult to answer. However, we do know that from the large number of booms made with the F-86 covering large areas of houses, barns, storerooms and the like, that comparatively little damage has actually been inflicted and then it was

confined largely to large plate glass windows with several isolated cases of cracked plaster or items knocked off of shelves, etc.

There are several known instances of high speed low level flight which did result in somewhat more severe damage. One instance was the F-100 flight demonstration for the press at Palmdale when 3 low passes made over the administration building with a diving supersonic start broke many 1/4 inch thick door and window glass plates and cracked the boards in the door frames. No one was injured but glass was reported to have been thrown far and hard.

Another case of more severe damage occurred during the speed record runs by the F-100 over the Salton Sea. Damage to 17 homes amounting to $3,600 was incurred in the form of broken doors, door frames, and windows.

Operational control of all supersonic aircraft is necessary in order to avoid serious property damage and to minimize the public nuisance aspects of the sonic boom. The Air Force and industry have already recognized this requirement and local operational procedures have been generally established which keep the aircraft at fixed distances from cities, populated areas, and other aircraft, as well as at high altitudes during supersonic flights. There is no known method of eliminating these shock waves. The aerodynamicist is working to reduce the wave drag of aircraft which will reduce the intensity of the waves but not avoid them.

There are no known cases of adverse physiological effects on humans although present day aircraft are capable of creating shocks with pressure jumps of about 40 pounds per square foot if they pass close enough, which is the range at about which human ears begin to rupture. The next item

of physical failure on the human body with increasing pressure jump is the lungs but the pressure required for damage to the lungs is about 5 atmospheres and this type of injury is therefore not anticipated.

From some of the reportings of boom effects, it appears that there is a tendency to panic if the source of the sound is unknown or not recognized. Some animals, however, appear to panic easily and some rather serious effects have been reported from various chicken farms.

To avoid these problems, the operational control previously mentioned appears to be the immediate answer along with a general education of the public regarding the sonic boom and its characteristics.

Military Application of the Boom

Now, what about possible military applications of the boom? It has been shown that extreme low altitude supersonic flight will result in high shock wave pressure jumps - at say, 100 feet altitude at Mach 1.5 pressures of the order of 100 pounds per square foot might be obtained. Wooden buildings would be especially susceptible to damage from strong shocks and in this pressure range serious physiological and psychological effects would be expected.

With respect to damaging other aircraft, it may be said that based on studies of blast damage to aircraft it is possible with present aircraft to obtain shock pressures in the range where certain types of damage are inflicted. However, because of the requirements of extreme nearness of the two aircraft, it presently appears that air-to-air offensive or defensive action is not practical.

Unanswered Problems

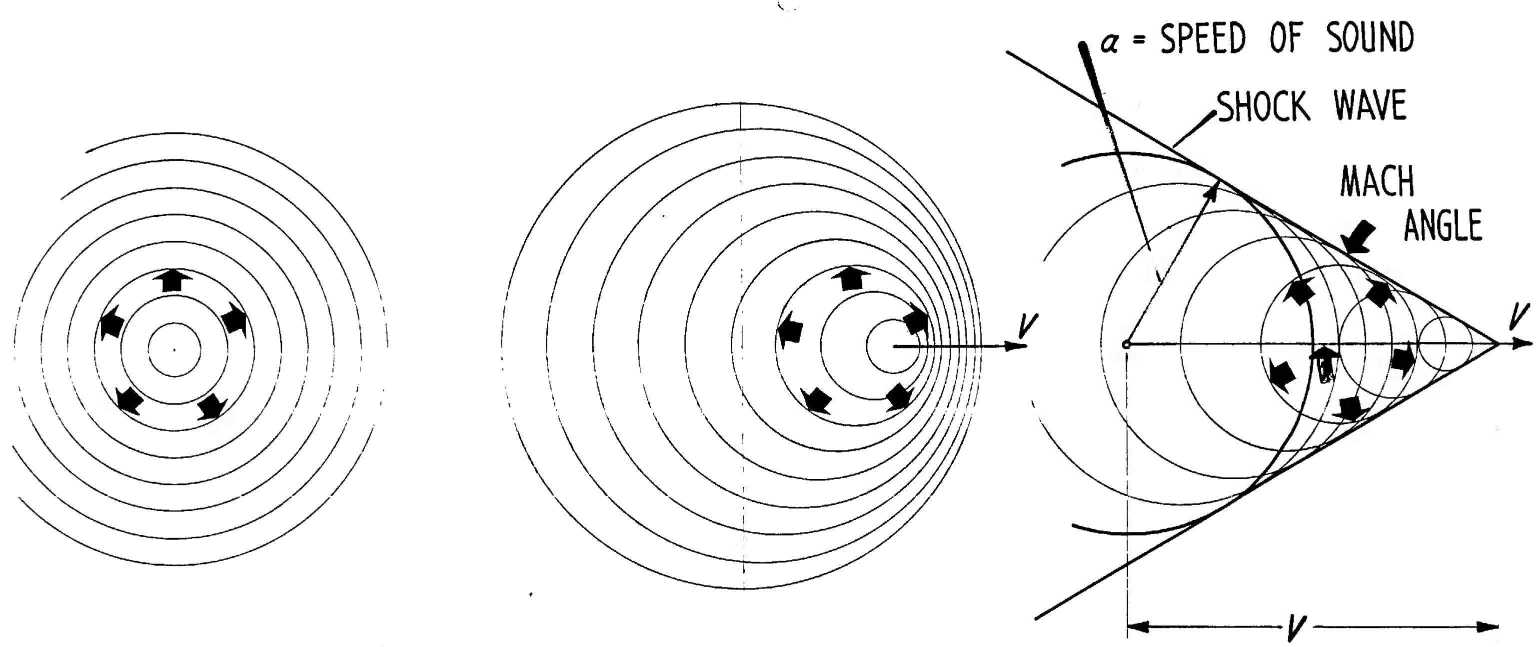
There remain quite a number of questions as yet unanswered. Past experience has indicated that although the pilot has a general knowledge of what ground area will receive the boom from his supersonic flight, it often happens that a remote or unexpected area is boomed. This is especially true for the case of maneuvering flight, where the aircraft is turning, pulling up, diving, etc., and it is now important that these transient effects be studied. There are also important questions, such as, "Is there an appreciable focusing effect due to turning flight which builds much higher than usual shock wave pressures on the inside of the turn?"; "What are the pressure characteristics of the shock waves generated by a formation of airplanes at supersonic speeds?" which must be answered since they could have quite a bearing regarding the application of shock waves as offensive or defensive weapons. Another question we ask is, "What can we do about the sonic boom?" The answer is that we know of no way to eliminate the booming shock waves completely. The aerodynamicists are working to decrease the wave drag of aircraft by using more slender wings and bodies, decreasing wing-body interference drag, properly locating engine air inlets, etc. Through these efforts the intensity of the wave may be decreased but it cannot be eliminated.

The question of "How do we prevent the sonic boom from doing damage to property or injury to humans?" can now only be answered by saying that we must apply operational control of all aircraft with supersonic speed capabilities and continue the study of the phenomenon in order that flight restrictions may be most intelligently established.

The physical size of the phenomenon involved and the many influencing variables make the experimental measurement problem quite complicated. It

is apparent that it is not practical to instrument the total ground area covered by the wave, but rather, instruments must be strategically located for providing data for correlation with theoretical results and predictions. The sonic boom problems and recently acquired test data are presently being evaluated at the Wright Air Development Center to clearly establish the particular areas in which further research and development efforts are desired. It is certainly necessary that the experimental study of this phenomenon be continued until all of the various facets involved have been thoroughly exploited and are clearly understood.

PROPAGATION OF SOUND



STATIONARY SOUND SOURCE

SOUND SOURCE MOVING AT CONSTANT SUBSONIC SPEED (M = 0.7)

SOUND SOURCE MOVING AT CONSTANT SUPERSONIC SPEED (M = 2.0)

Figure 1 PROPAGATION OF DISTURBANCE

RELATION OF OVERALL SOUND PRESSURE LEVEL TO HEARING SENSATION

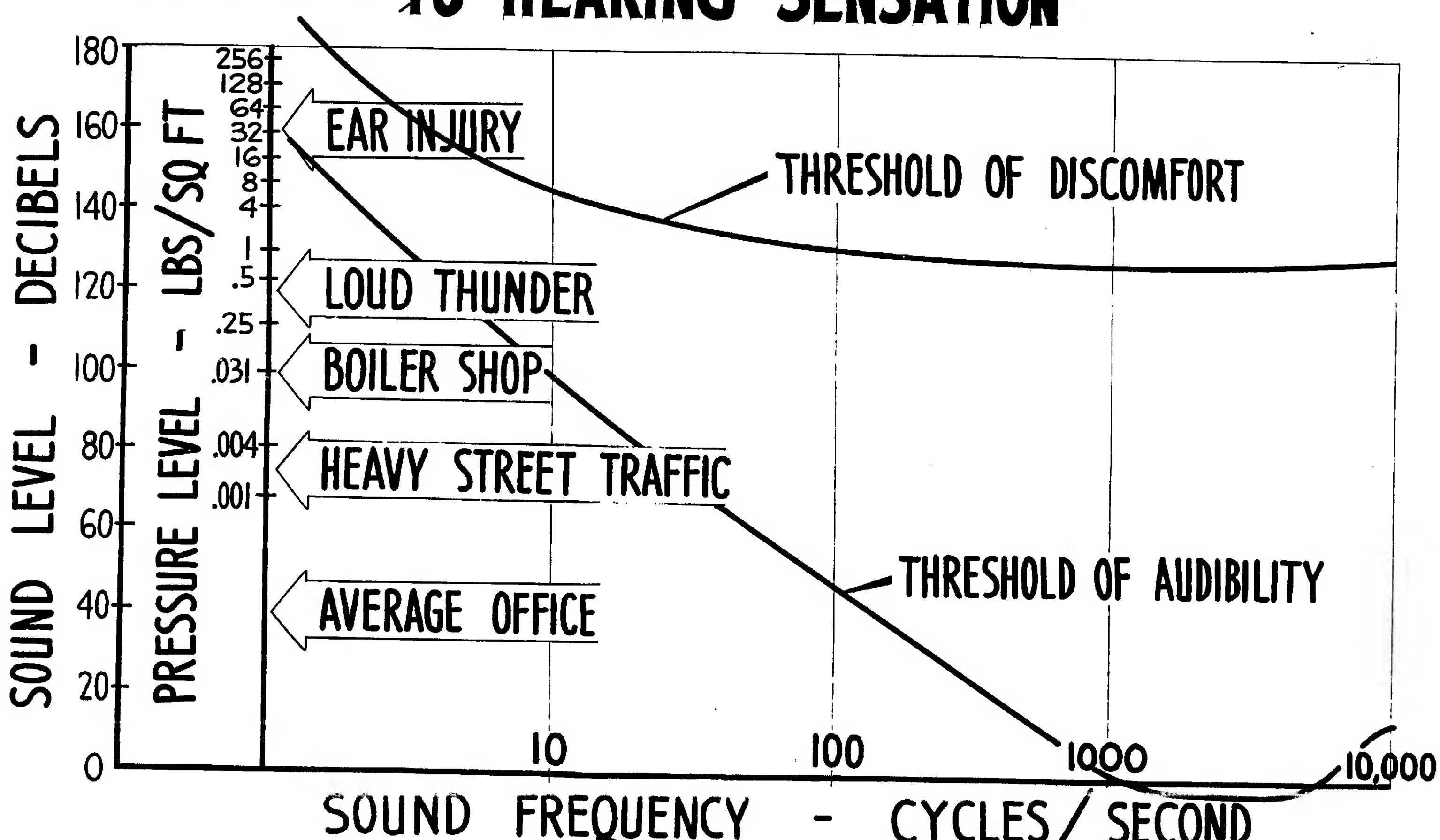


Figure 2 SOUND FREQUENCY CYCLES SEC

SHOCK WAVE PATTERNS IN STEADY STATE AND TRANSIENT PHENOMENON

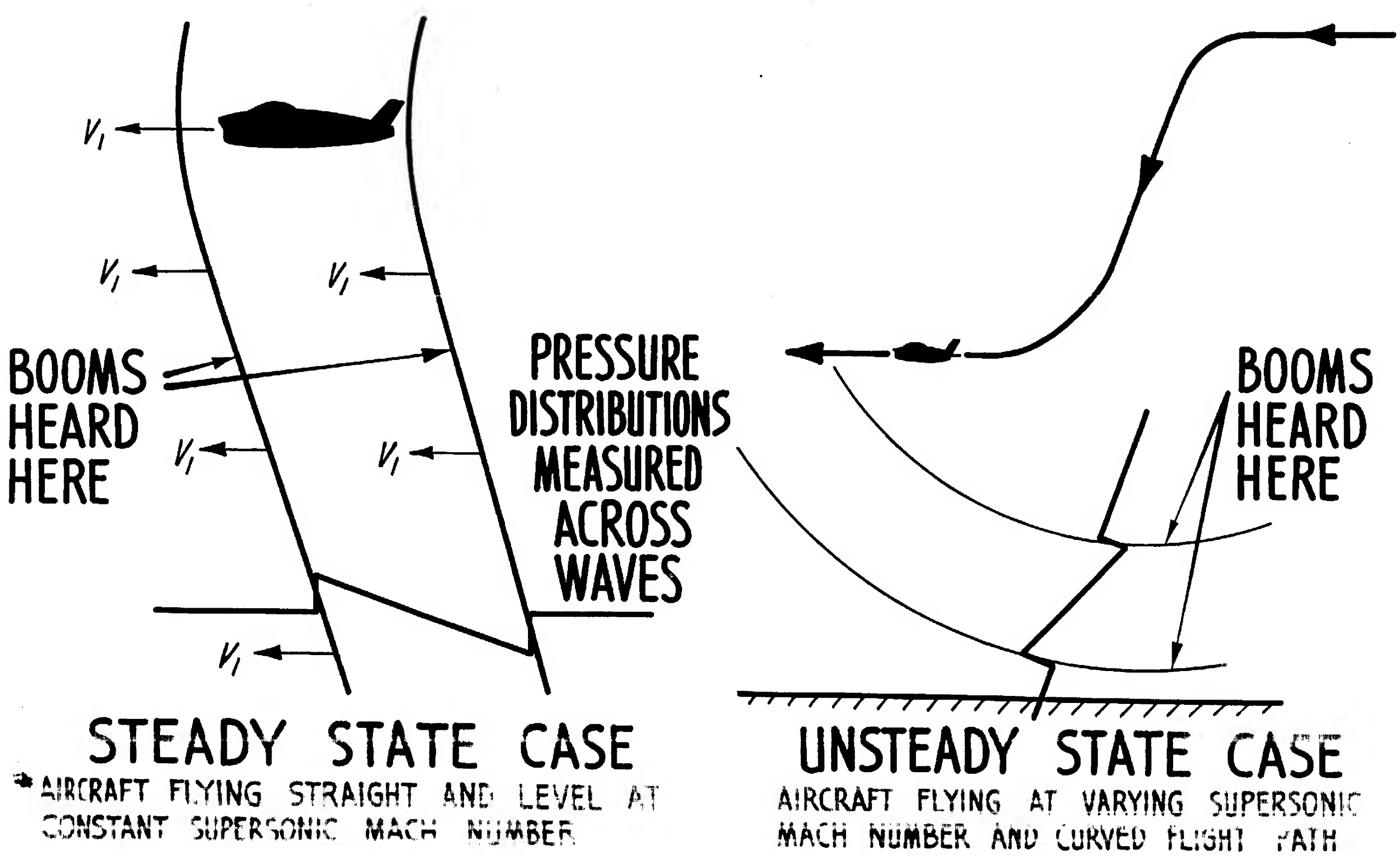


Figure 3 ILLUSTRATING SHOCK WAVE PATTERNS IN STEADY STATE AND TRANSIENT PHENOMENON

THEORETICAL PRESSURE RISE ACROSS BOW WAVE
EFFECT OF MACH NUMBER

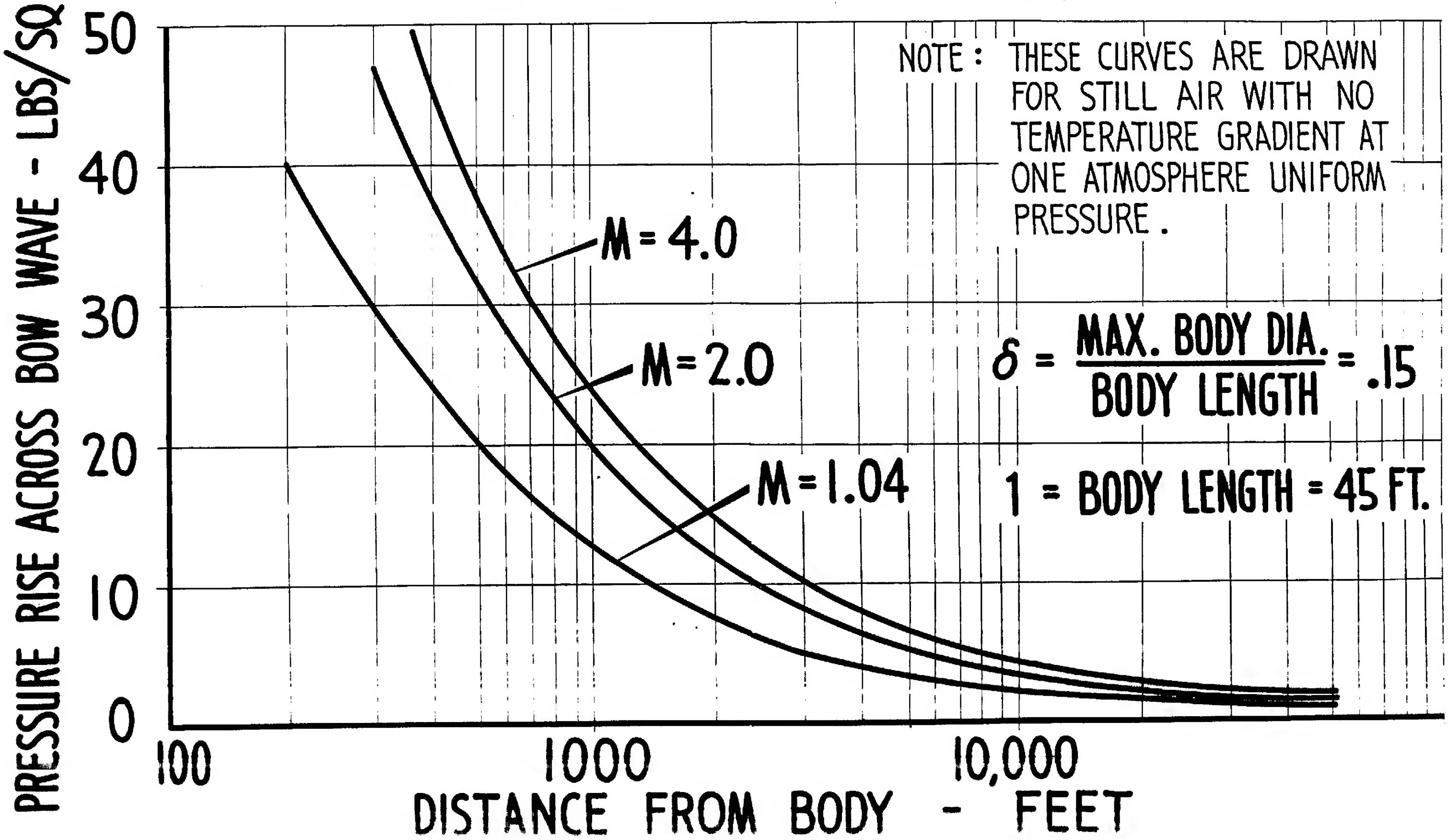


Figure 4 THEORETICAL PRESSURE RISE ACROSS BOW WAVE - EFFECT OF MACH NUMBER

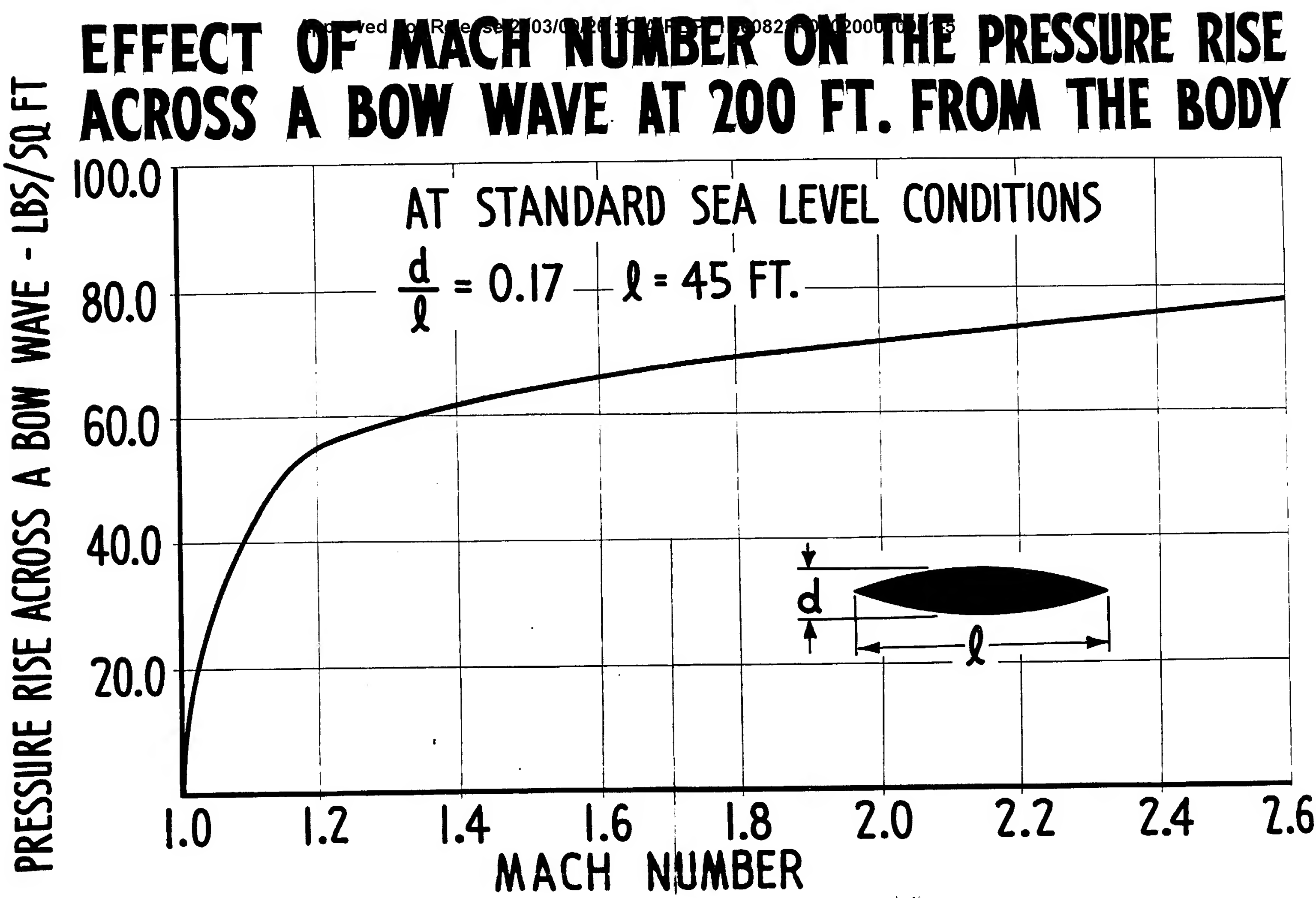


Figure 5 EFFECTS OF MACH NUMBER ON THE PRESSURE RISE ACROSS A BOW WAVE AT 200 FEET FROM THE BODY AT STANDARD SEA LEVEL CONDITIONS

THEORETICAL PRESSURE RISE ACROSS BOW WAVE
EFFECT OF BODY SIZE

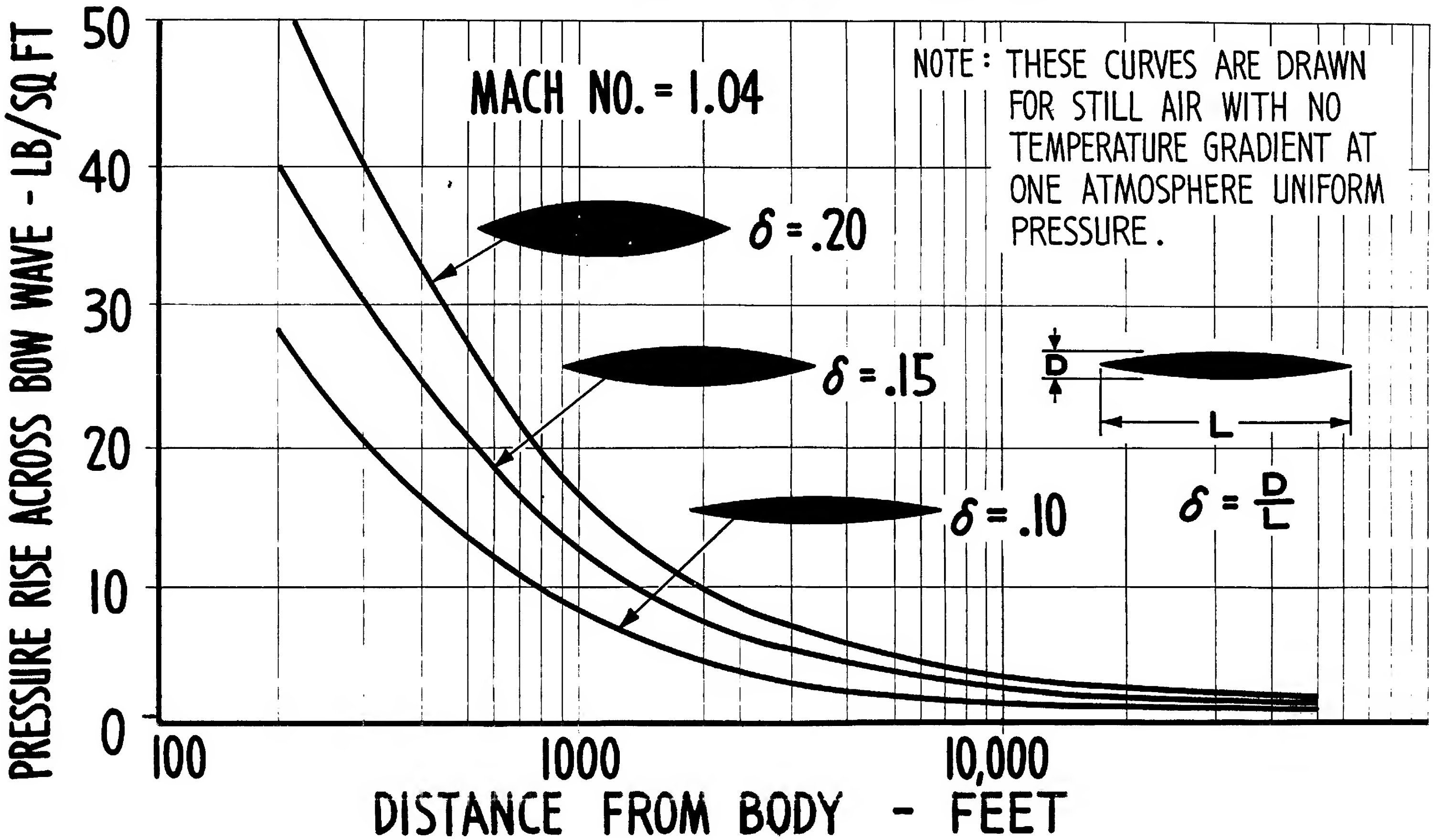


Figure 5 THEORETICAL PRESSURE RISE ACROSS BOW WAVE - EFFECT OF BODY SIZE

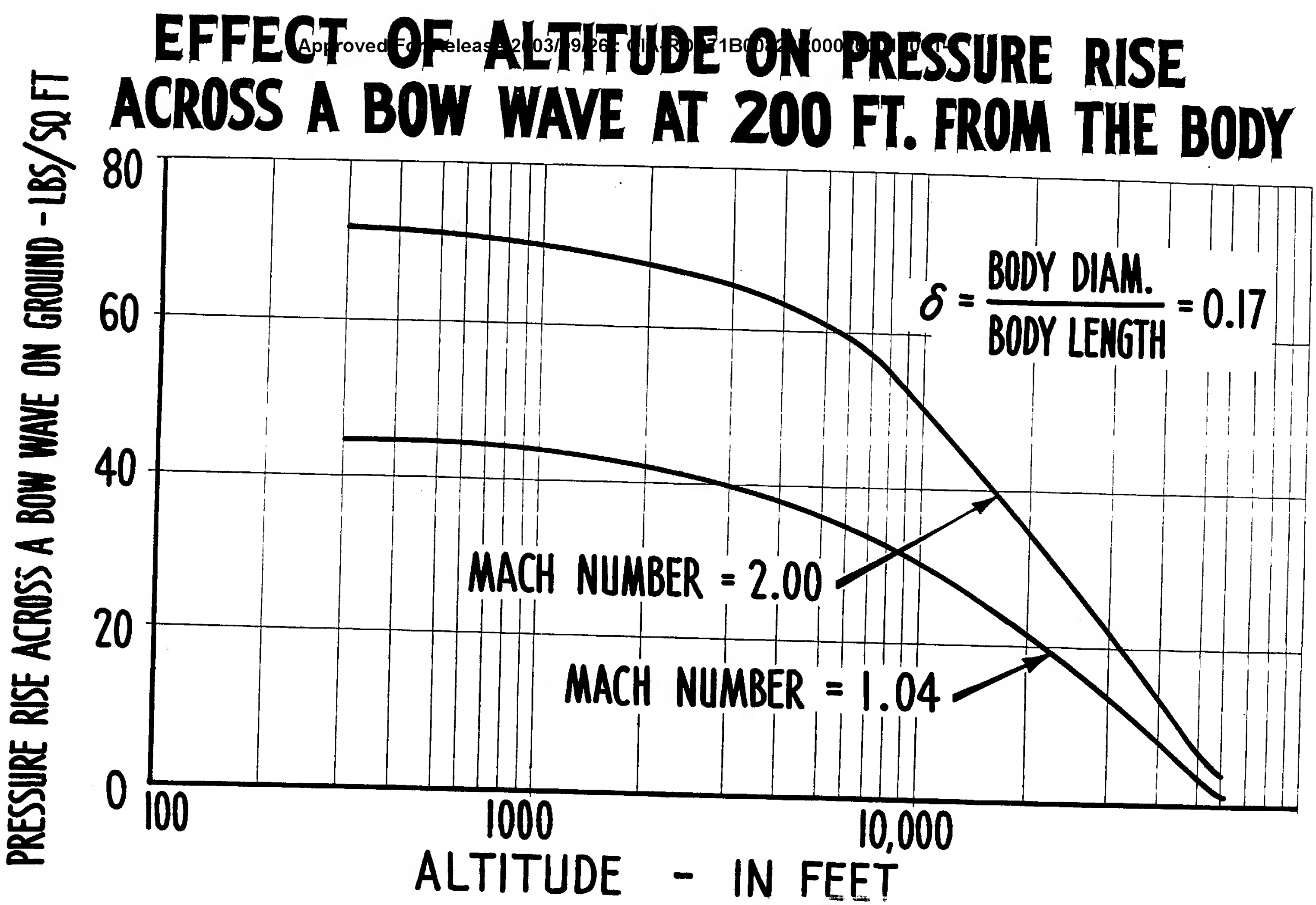


Figure 7 EFFECT OF ALTITUDE ON PRESSURE RISE ACROSS A BOW WAVE AT 200 FEET FROM THE BODY

EFFECTS OF ATMOSPHERIC TEMPERATURE GRADIENTS ON SHOCK WAVES FOR THE STEADY STATE CASE

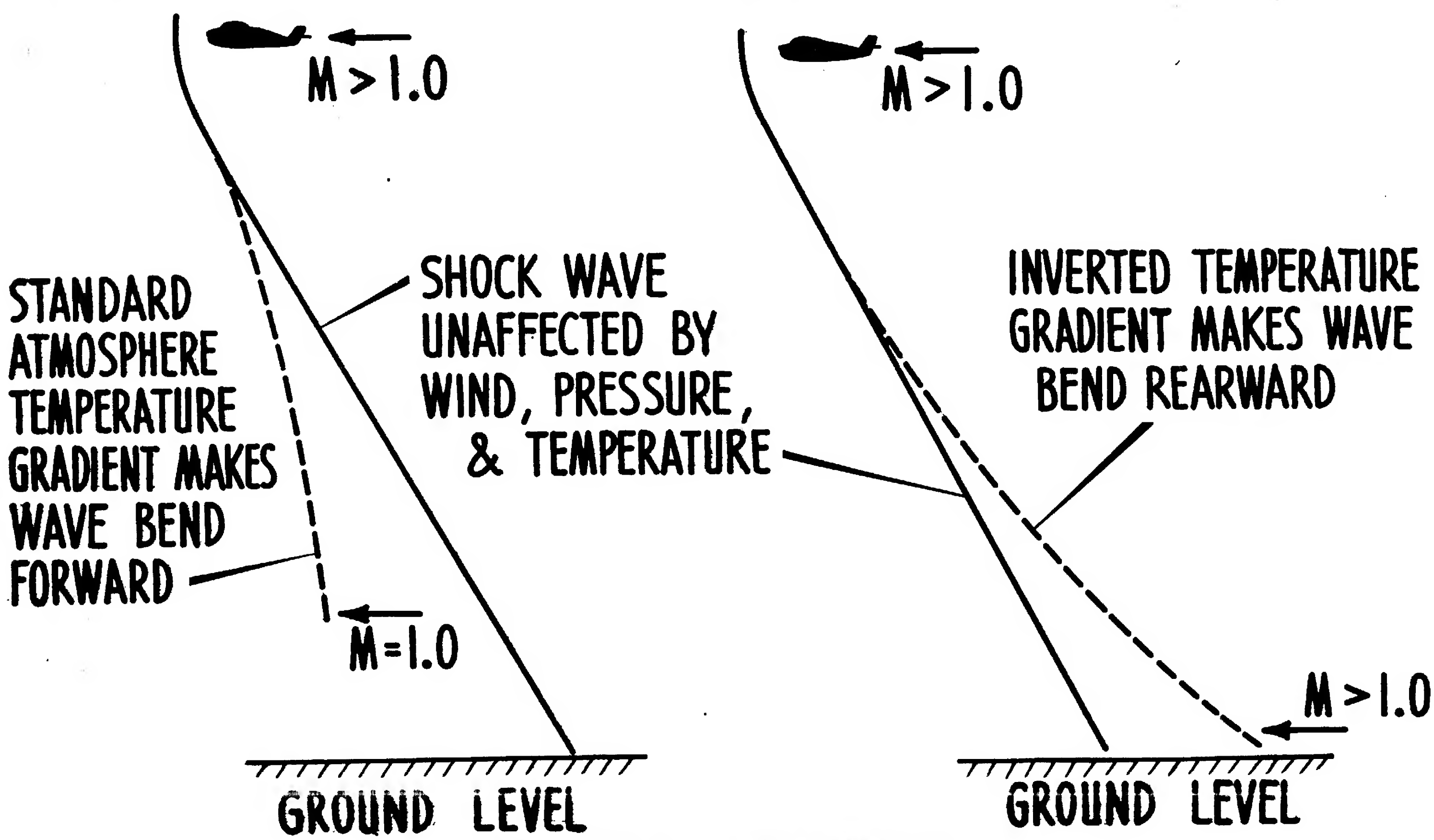


Figure 8 EFFECTS OF ATMOSPHERIC TEMPERATURE GRADIENTS ON SHOCK WAVES FOR THE STEADY STATE CASE

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EFFECTS OF ATMOSPHERIC WIND GRADIENTS ON SHOCK WAVES FOR THE STEADY STATE CASE

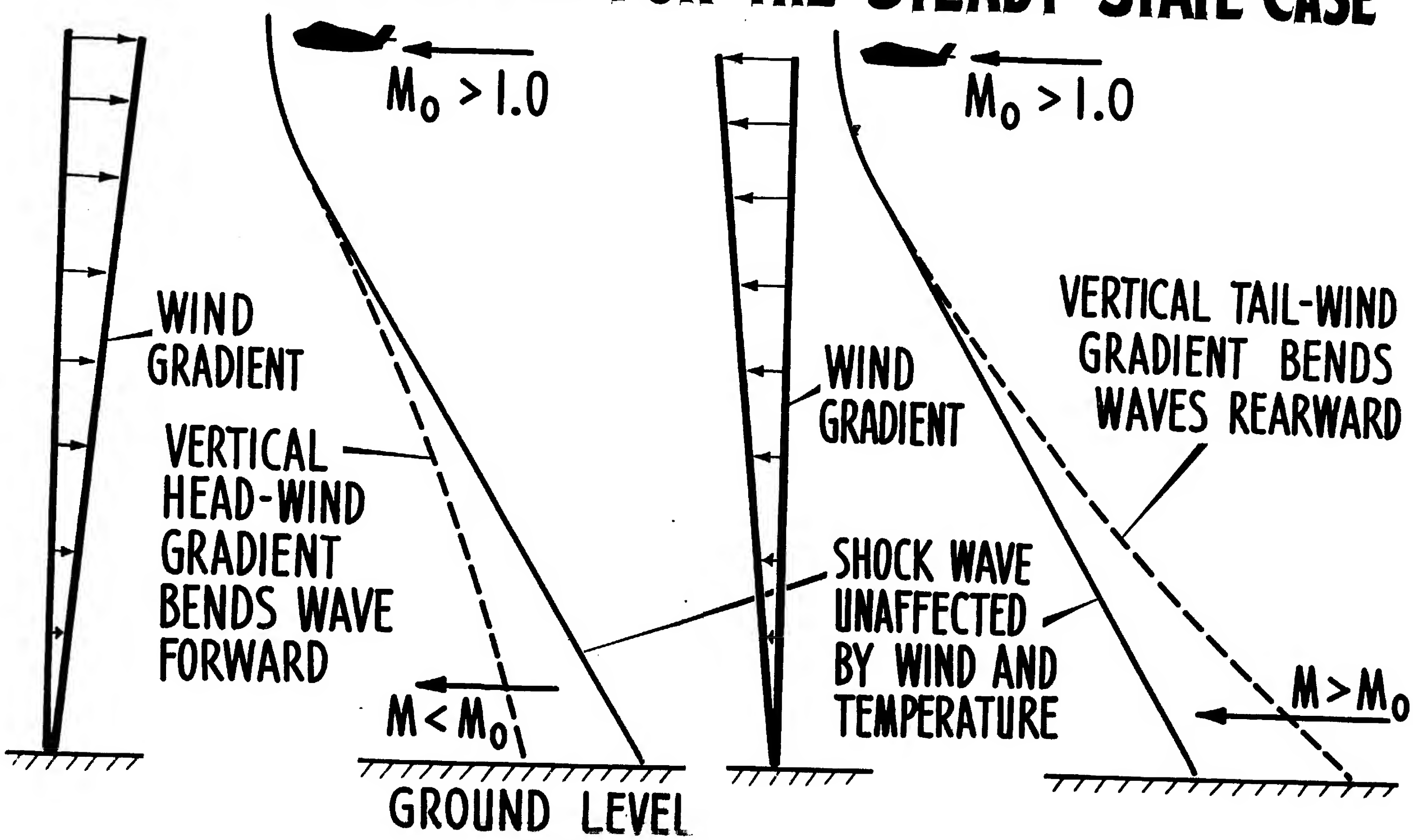

Figure 9 EFFECTS OF ATMOSPHERIC WIND GRADIENTS ON SHOCK WAVES FOR THE STEADY STATE CASE

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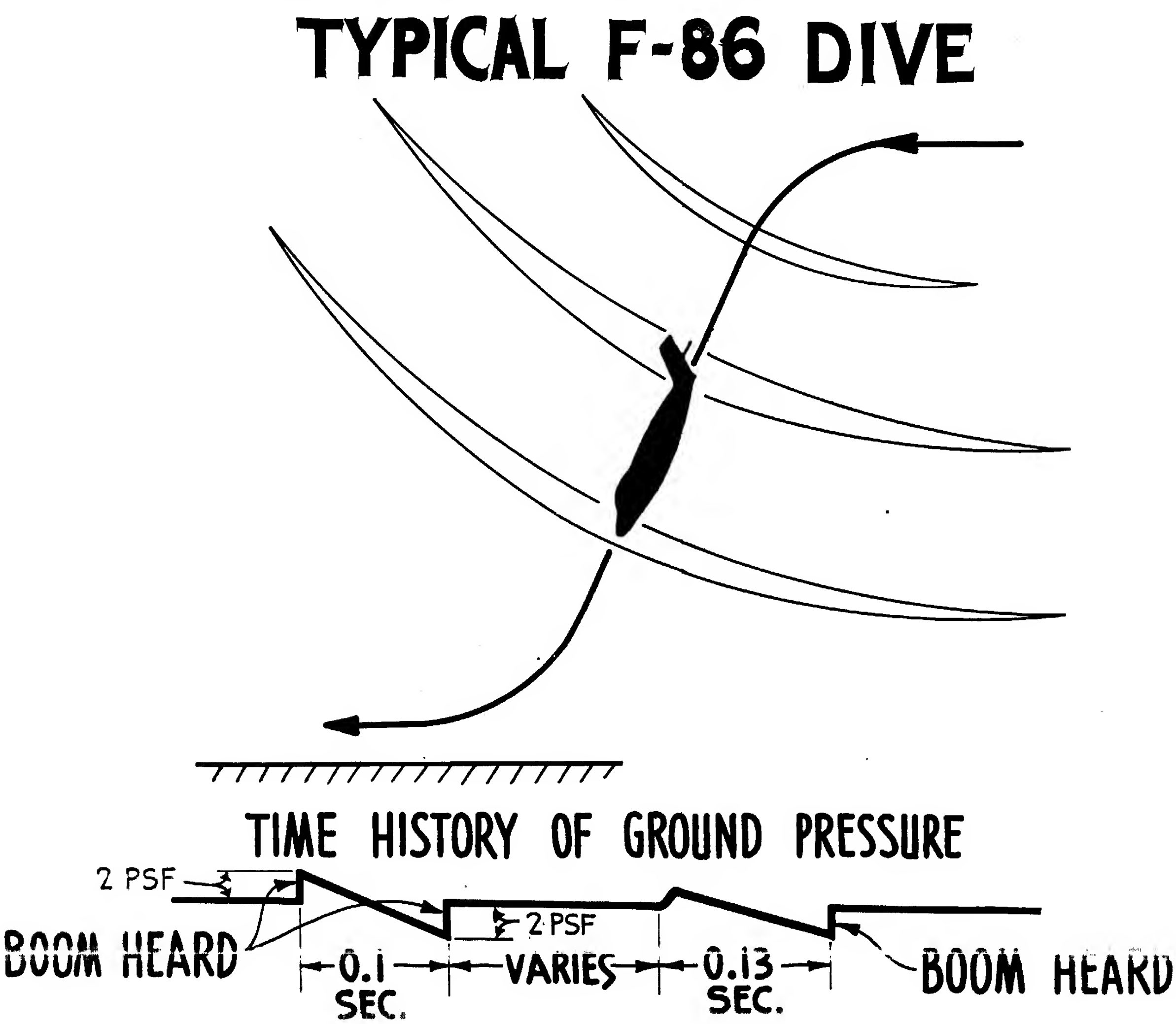

Figure 10 TYPICAL F-86 DIVE

EXPERIMENTAL MEASUREMENT OF PRESSURE JUMP

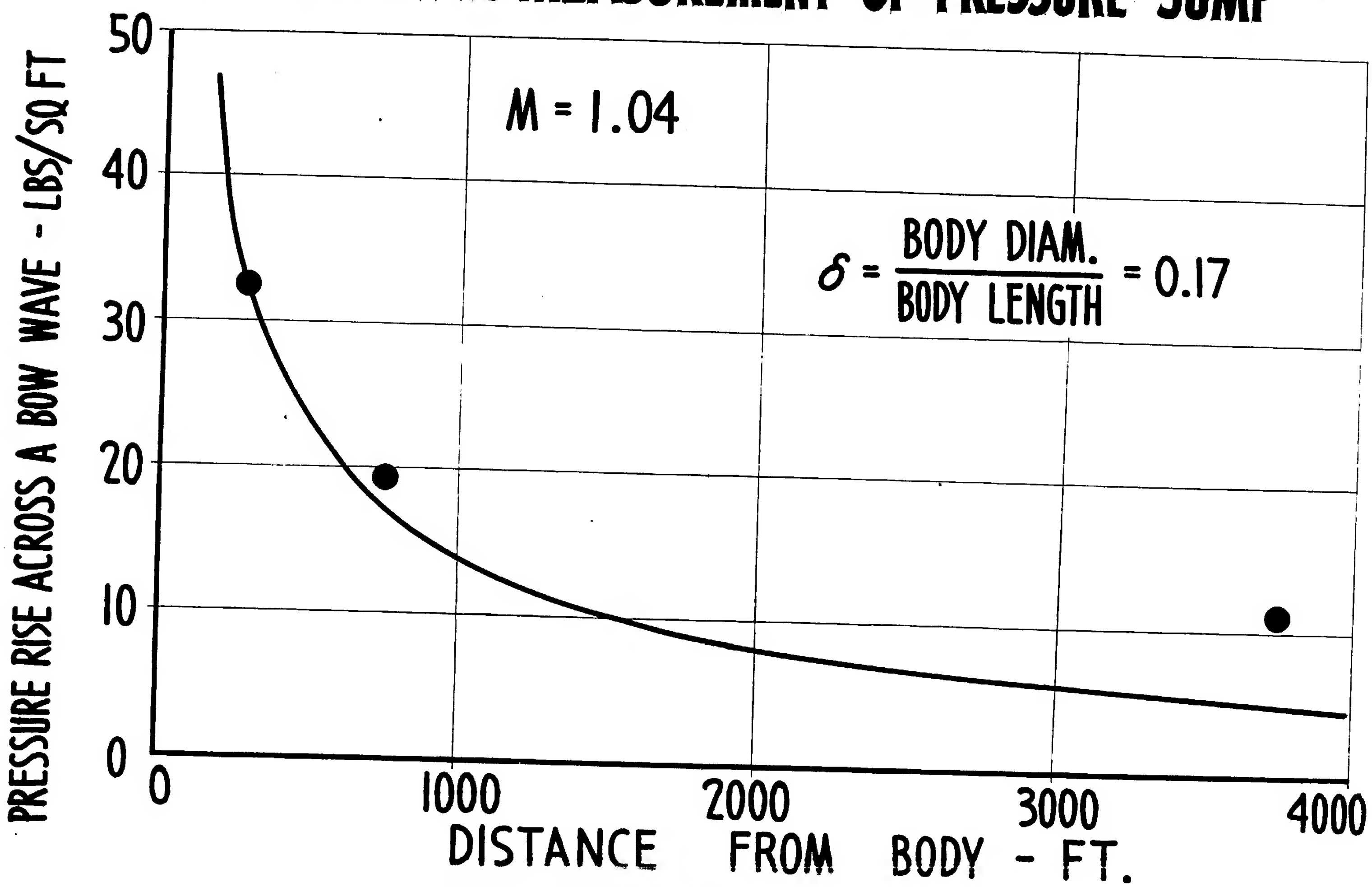

Figure 11 EXPERIMENTAL MEASUREMENT OF PRESSURE JUMP

ALTITUDE-MACH NUMBER LIMITS FOR CREATING SONIC BOOM IN LEVEL FLIGHT

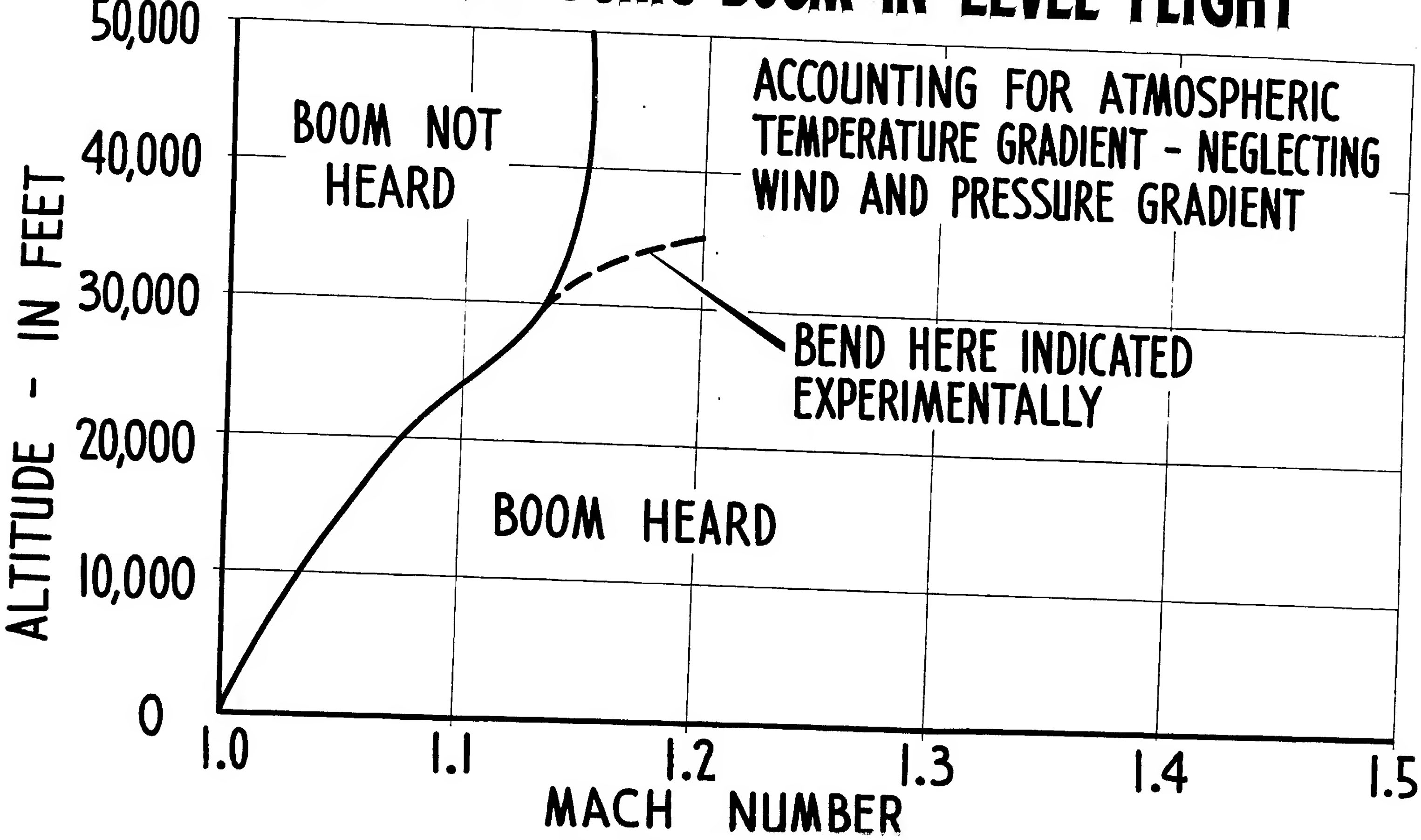

Figure 12 ALTITUDE - MACH NUMBER LIMITS FOR CREATING SONIC BOOM IN LEVEL FLIGHT - ACCOUNTING FOR ATMOSPHERIC TEMPERATURE GRADIENT - NEGLECTING WIND AND PRESSURE GRADIENT

ROUGH EXPERIMENTAL OBSERVATION OF SPREAD OF SONIC BOOM

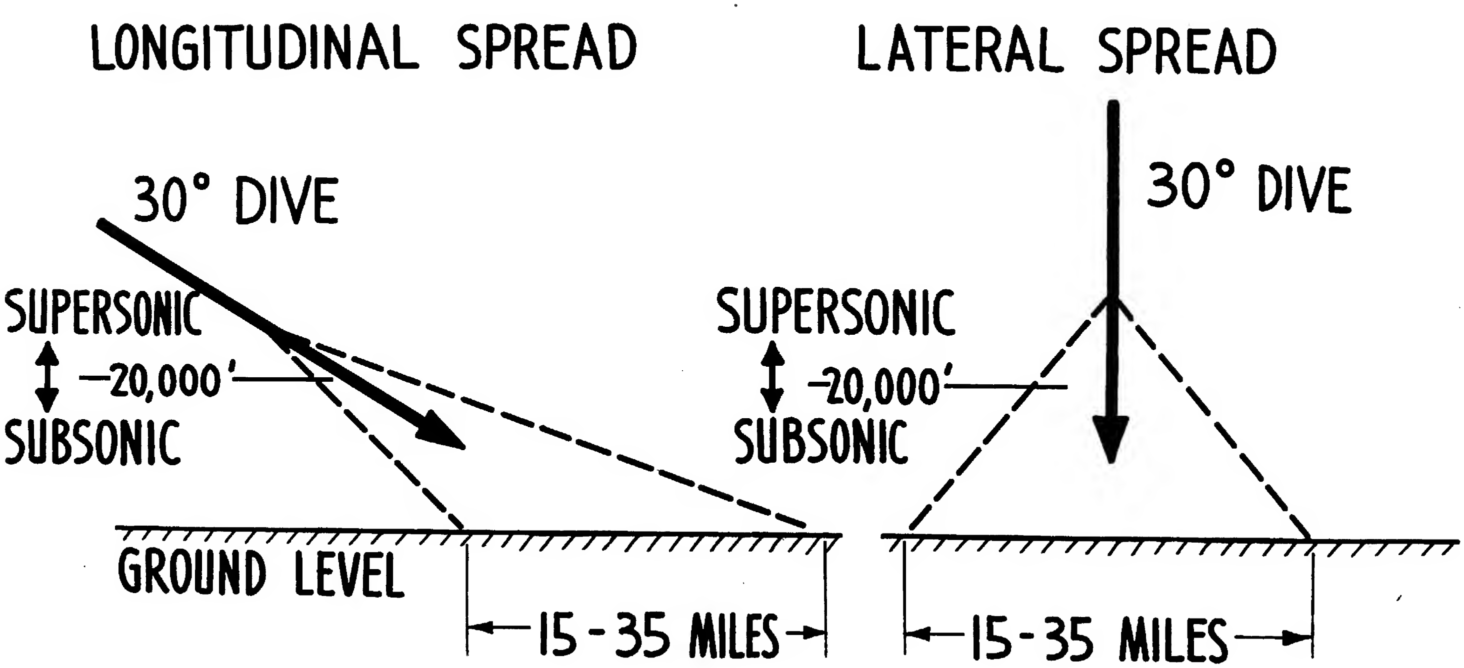

Figure 13 ROUGH EXPERIMENTAL OBSERVATION OF SPREAD OF SONIC BOOM